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ENGINE/AIRFRAME
COMPATIBILITY STUDIES
FOR SUPERSONIC CRUISE AIRCRAFT

FINAL REPORT

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FOREWORD

This document presents the results of a contract study entitled "Engine/ Airframe Compatibility Studies for Supersonic Cruise Aircraft" performed for NASA by the Douglas Aircraft Company, McDonnell Douglas Corporation.

The NASA technical monitor for the study was F. E. McLean, Advanced Supersonic Technology Office, Langley Research Center, Hampton, Virginia.

This study program was under the overall direction of R. D. FitzSimmons, Director, Advanced Supersonic Transport. The Technical Manager was W. T. Rowe. This report consists of results of in-depth analyses of a supersonic transport configuration with alternate engines (mini-bypass turbojet, duct burning turbofan, and variable cycle) integrated in place of a dry turbojet engine.

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SYMBOLS AND ABBREVIATIONS

A.C., a.c.	Aerodynamic Center
A_c	Inlet Capture Area
A_o	Freestream Capture Area
A_o/A_c	Mass-flow Ratio
$A_o/A_c)_{bleed}$	Mass-flow Ratio of Inlet Bleed Flow
ARROW	Automatic Re-analysis and Redesign for Optimum Weight
Aux	Auxiliary
$A_{9.1}$	Nozzle Exit Area
A_{10}	Nozzle Reference Area
BPR	Bypass Ratio
BTU	British Thermal Unit
c	Local Wing Chord
$\bar{c}$	Mean Aerodynamic Chord
^oC	Temperature-Celsius
C_D	Drag Coefficient = $\frac{D}{q_o S_{ref}}$
C.G., c.g.	Center of gravity
$C_{L\alpha_W}^{E}$	Lift Curve Slope of Elastic Wing
$C_{L\alpha_W}^{R}$	Lift Curve Slope of Rigid Wing
$C_{L\alpha}^{E/R}$	Ratio of Elastic to Rigid Lift Curve Slope
cm	Centimeter
CPR	Cycle Pressure Ratio
D	Drag, Diameter
D_{AFT}	Afterbody Drag

DAC	Douglas Aircraft Company
DH	Duct Heating
DOC	Direct Operating Cost
EAS	Equivalent Air Speed
ECS	Environmental Control System
EGT	Exhaust Gas Temperature
EPNdB	Effective Perceived Noise Levels in Decibels
FT.	Feet
°F	Temperature- Fahrenheit
F_G	Gross Thrust
F_N	Net Thrust
FAR Part 36	Federal Air Regulations for Noise
FORMAT	Fortran Matrix Abstraction Technique
FPR	Fan Pressure Ratio
fps	Feet per Second
g	Acceleration of Gravity
GE	General Electric
gm	Gram
HP	Horsepower
HPC	High Pressure Compressor
HPT	High Pressure Turbine
HR	Hour
in	Inches
J	Joules
°K	Temperature-Kelvin
k	Kilo
KEAS	Knots Equivalent Air Speed

kg	Kilograms
km	Kilometers
kW	Kilowatts
L	Lift
LB.	Pounds
L/D	Lift to Drag Ratio
LP	Low Pressure
m	meters
M	Mach Number
MALS	Matrix Aeroelastic Loads System
MAPES	Mass Properties Estimation System
N	Newtons
NASA	National Aeronautics and Space Administration
N.MI., n.mi.	Nautical Miles
P&WA	Pratt and Whitney Aircraft
PNdB	Perceived Noise-Decibels
PNL	Perceived Noise Level
PPS	Pounds per Second
P_{amb}	Ambient Pressure
P_o	Sea Level Pressure, 2116.2 LB/FT^2 (10.1325 N/cm^2)
P_{to}	Freestream Total Pressure
P_{t2}	Average Compressor Face Total Pressure
P_{t2}/P_{to}	Inlet Total Pressure Recovery
q	Freestream Dynamic Pressure, $1/2 \rho V^2$
sec	Seconds
SFC	Specific Fuel Consumption
SL	Sea Level
SLS	Sea Level Static

Std	Standard
TF	Turbofan
TIT	Turbine Inlet Temperature
TJ	Turbojet
V	Velocity
V_j	Jet Velocity
T_{41}	GE Design Rotor Inlet Temperature
T_{amb}	Static Temperature
T_o	Sea Level Static Temperature, 518.7°R (288.16°K)
T_{t2}	Inlet Total Temperature
VCE	Variable Cycle Engine
W	Weight
W_a	Engine Airflow
WAT2	Corrected Inlet Airflow
W_f	Engine Fuel Flow
$\Delta X_{a.c.}$	Change in Aerodynamic Center Location
"	Inches
%	Percent
δ_{amb}	Pressure Ratio, P_{amb}/P_o
δ_{t2}	Pressure Ratio, P_{t2}/P_o
θ_{t2}	Temperature Ratio, T_{t2}/T_o
η	$C_{L\alpha}^{E/R}$

INTRODUCTION

During 1973, the Douglas Aircraft Company (DAC), McDonnell Douglas Corporation conducted technology assessment studies in support of the NASA program of providing an updated technology base from which an advanced supersonic cruise aircraft can be produced with a high probability of success. The initial phase included an assessment of gains available through application of advanced technologies in aerodynamics, propulsion, acoustics, structures, materials, and active controls. The second phase encompassed an assessment of the potential market and range requirements as well as economic factors including payload, speed, airline operating costs, and airline profitability. A third phase consisted of identifying and completing the conceptual design of a baseline aircraft which could be used to assess technology requirements in detail. These studies culminated with a major technology assessment detailing which technology achievements the U.S. industry and government should emphasize in the near term in order to provide a proven technology base for program initiation.

The results of the above noted studies and technology assessments are reported in a NASA contract study report Douglas MDC J4394 (Volumes I thru IV), "Studies of the Impact of Advanced Technologies Applied to Supersonic Transport Aircraft", dated September, 1973.

During the period of the above noted studies, the major U.S. engine manufacturers were engaged in NASA funded studies to define conceptual engines for application to an efficient and quiet advanced supersonic cruise aircraft. The engine types studied included the dry turbojet, afterburning turbojet, duct heating turbofan, afterburning turbofan, and variable cycle engines. Through initial screening of these data, DAC concluded that the dry turbojet was the most practical and efficient engine for the time period considered, although considerable noise suppression was required to make it acceptable. It provided the lowest operating cost airplane design. This became the DAC baseline airplane configuration.

Subsequent to this baseline definition, additional NASA funded engine cycle refinement studies by the engine manufacturers have been made available to the airframe manufacturers. Some of these engine designs offered potential advantages over the dry turbojet, however, a reasonable assessment of engine comparisons for an advanced supersonic cruise vehicle cannot be confidently done without detail analysis and technical integration with an airframe. Since development of an engine seems to be the pacing item for any new supersonic transport program, it is most important to define the correct cycle at the earliest possible date. With close coordination between DAC and the engine manufacturers such definition can be accomplished. DAC undertook the studies described herein with the understanding that the engine companies most promising concepts would be evaluated, their data used exclusively, and their maximum technology projections incorporated at face value. DAC provided the design team and expertise to accomplish the complex engine/airframe integration tasks including sizing each engine and doing a sophisticated airplane integration.

The effort described in this report includes the analysis and synthesis leading to the selection and integration of a "best" to date available engine from each of the turbojet, duct heating turbofan and variable cycle engine families. The resulting performance and acoustics data are presented as compared with the earlier established DAC conceptual baseline configuration.

SUMMARY

Engine/airframe compatibility studies have been completed utilizing the DAC advanced supersonic transport point design as the baseline for comparison. After analysis of many options, a specific engine design was selected for each of three types of engine cycles and a careful engine airframe integration study completed for each relative to the point design airplane. The engines selected for detail study are as follows:

- Mini-bypass turbojet the GE P7 engine
- Duct heating turbofan the P&WA 501D engine
- Valved variable cycle the P&WA 302B engine

These engines were selected as the best available within the time frame of this study and are reported on as offered without any technology normalizing between supplying engine companies or within a specific company.

This effort has been accomplished under NASA contract No. 1-13229. This report fulfills the requirement for a final summary report on this effort.

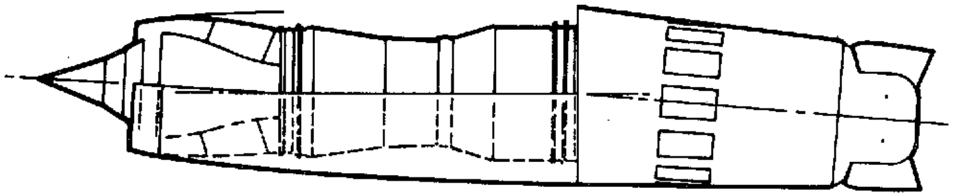
These studies entail a preliminary design process which integrates the technical variations necessary to size the candidate engines, define the nacelle and airplane geometry, determine new aerodynamic, propulsion and weight efficiencies, and then assess the resulting performance and acoustics characteristics as compared with the base point design airplane [750,000 lb. (340,194 kg) takeoff groww weight, 10,000 ft.^{2} (929m^2) wing area, and 273 passengers]. The initial engine sizing constraint used for the study is that each study engine produce a takeoff thrust at 0.3 Mach equivalent to the reference airplane engine with sideline and takeoff/cutback noise not to exceed FAR Part 36. This sizing is later validated by determining the engine size which provides best range.

The method used in the acoustics evaluation of the engines during the study is to calculate the unsuppressed source noise knowing the engine details plus the gas flow data provided for the particular engine operating condition. These inputs are based on engine cycle data supplied by the engine manufacturers, as part of their contract efforts with NASA Lewis. Initial jet noise suppression values applied to the unsuppressed jet noise levels are based on suppressor characteristics supplied by each engine manufacturer for his engine cycle. Later information from one engine manufacturer indicated a reduction in suppression levels based on recent test data corrected for forward flight effects. As a result, information is included considering the DAC baseline configuration nozzle/suppressor/reverser exhaust system, including effects of forward flight on that particular engine.

It is estimated that the approach noise levels are of the same order of magnitude as those for the baseline configuration and less than FAR Part 36 requirements.

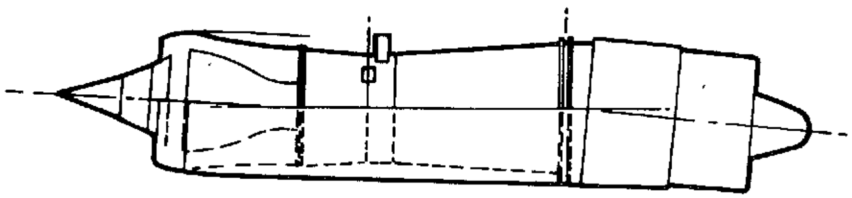
The technical analysis, configuration descriptions, and study results are presented in subsequent sections of this report by technology or multi-technology area of responsibility for each of the three specific engines studied.

A summary chart illustrating the resulting sizes required for the various engines studied is presented in Figure A-1. A summary of takeoff performance is shown in Table A-1, and a noise summary is provided in Table A-2. Specific fuel consumption data are summarized in Table A-3 and relative ranges in Figure A-2. The variation in operators' weight, L/D and range with engine size for the study engines is summarized in Figure A-3. The engine sizes identified in Figure A-3 are the minimum sized engines meeting the initial sizing constraints. The mini-bypass turbojet shows near-optimum range. The duct heating turbofan cannot be reduced in size since the temperature of the fan stream impacting the suppressor



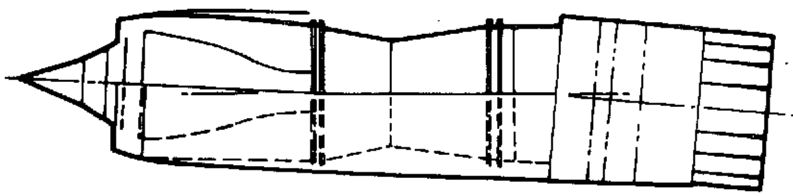
BASELINE TURBOJET ENGINE
AIRFLOW 773 LB/SEC (351 kg/SEC)

WEIGHT (ENGINE AND NOZZLE)	16,982 LB (7679 kg)
LENGTH (INTAKE FACE TO NOZZLE)	438.5 IN. (11.1 m)
MAX DIAMETER	97.0 IN. (2.5 m)
SLS THRUST	73,200 LB (326 kN)



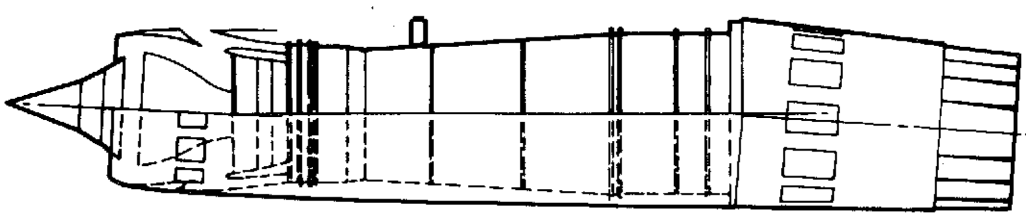
MINI-BYPASS ENGINE
AIRFLOW 782 LB/SEC (355 kg/SEC)

WEIGHT (ENGINE AND NOZZLE)	14,413 LB (6538 kg)
LENGTH (INTAKE FACE TO NOZZLE)	359.5 IN. (9.1 m)
MAX DIAMETER	81.0 IN. (2.06 m)
SLS THRUST	74,700 LB (332 kN)



DUCT HEATING TURBOFAN ENGINE
AIRFLOW 875 LB/SEC (397 kg/SEC)

WEIGHT (ENGINE AND NOZZLE)	12,200 LB (5543 kg)
LENGTH (INTAKE FACE TO NOZZLE)	358.5 IN. (9.1 m)
MAX DIAMETER	86.8 IN. (2.2 m)
SLS THRUST	70,000 LB (314.5 kN)



VARIABLE CYCLE ENGINE 302B
AIRFLOW 1003 LB/SEC (455 kg/SEC)

WEIGHT (ENGINE AND NOZZLE)	19,575 LB (8879 kg)
LENGTH (INTAKE FACE TO NOZZLE)	496.89 IN. (12.6 m)
MAX DIAMETER	100.23 IN. (2.5 m)
SLS THRUST	58,050 LB (258 kN)

FIGURE A-1. ENGINE SUMMARY

TABLE A-1
TAKEOFF SUMMARY

	REFERENCE TURBOJET -5A	MINI-BYPASS -5B	DH/TF -5C	VCE -5D
FIELD LENGTH (FT)	10,700 (3261 m)	10,850 (3307 m)	11,200 (3383 m)	11,000 (3350 m)
HEIGHT AT 3.5 N MI (FT)	1,256 (383 m)	1,292 (394 m)	1,268 (386 m)	1,225 (373 m)

TABLE A-2
NOISE SUMMARY

FAR PART 36 NOISE LEVELS/FAR PART 36 NOISE REQUIREMENTS, EPNdB

ENGINE TYPE	SIDELINE	CUTBACK
DAC DRY TURBOJET (-5A)	104/–4	105/–3
GE P7 WITH 15 EPNdB SUPPRESSOR	102/–6	107/–1
GE P7 WITH 12 EPNdB SUPPRESSOR (-5B)	105/–3	110/+2
GE P7 WITH DAC SUPPRESSOR	106/–2	110/+2
DUCT HEATING TURBOFAN (-5C)	108/0	108/0
VARIABLE CYCLE 302B (-5D)	107/–1	106/–2

TABLE A-3
ENGINE SFC SUMMARY

	REFERENCE TURBOJET	P-7	501D	302B
CLIMB SFC, UNINSTALLED (1.57M, 40K FT)	1.165	1.182	1.350	1.380
CLIMB SFC, INSTALLED (1.57M, 40K FT)	1.258	1.227	1.546	1.593
SUBSONIC CRUISE, UNINSTALLED (0.93M, 30K FT)	1.150	1.075	1.050	1.050
SUBSONIC CRUISE, INSTALLED (0.93M, 30K FT)	1.420	1.195	1.320	1.300
SUPERSONIC CRUISE, UNINSTALLED (2.2M, AVG CR ALT)	1.270	1.270	1.455	1.350
SUPERSONIC CRUISE, INSTALLED (2.2M, AVG CR ALT)	1.376	1.348	1.644	1.496
HOLD, UNINSTALLED (0.55M, 15K FT)	1.350	1.165	0.980	0.960
HOLD, INSTALLED (0.55M, 15K FT)	1.440	1.229	1.110	1.130

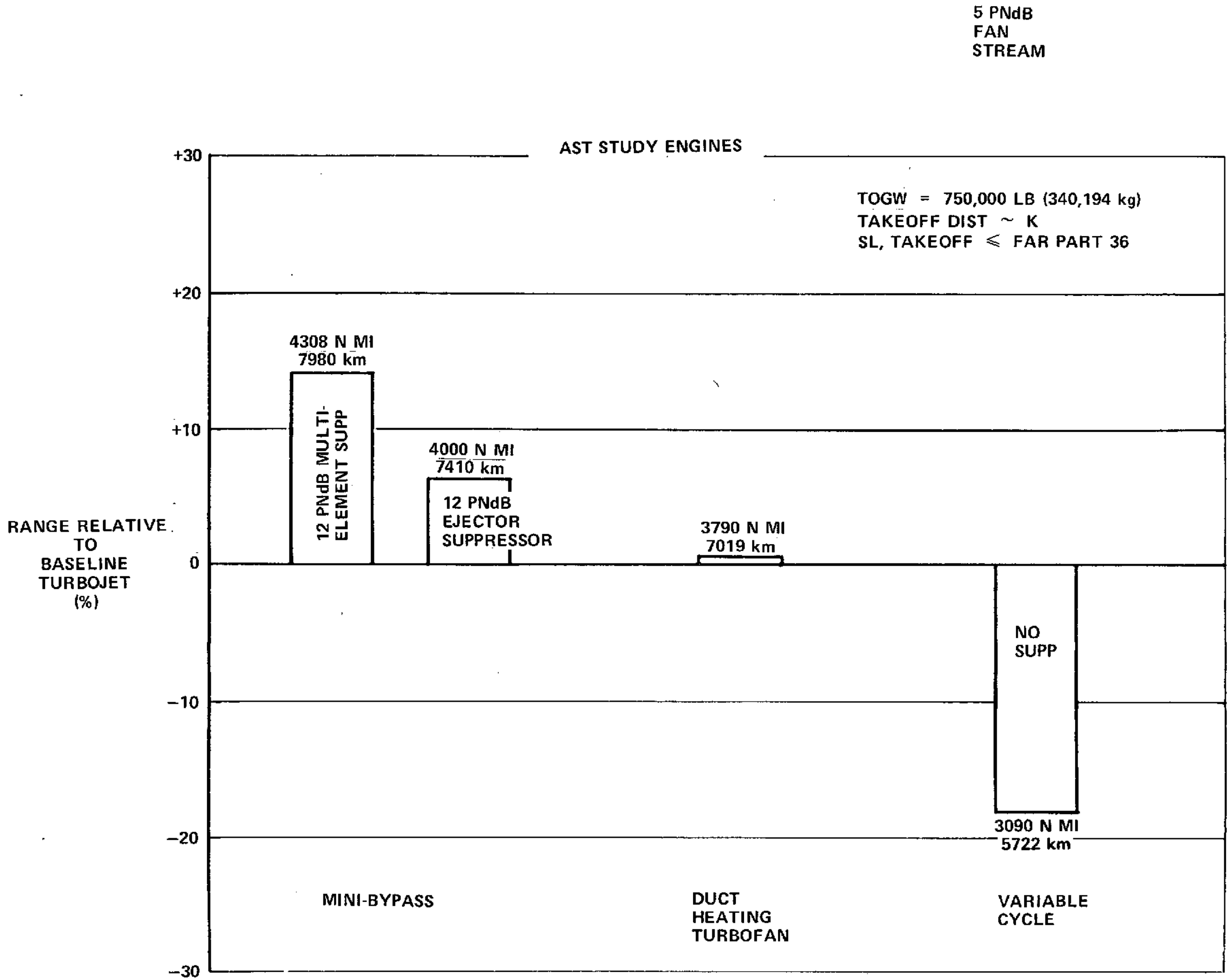


FIGURE A-2. RANGE COMPARISON

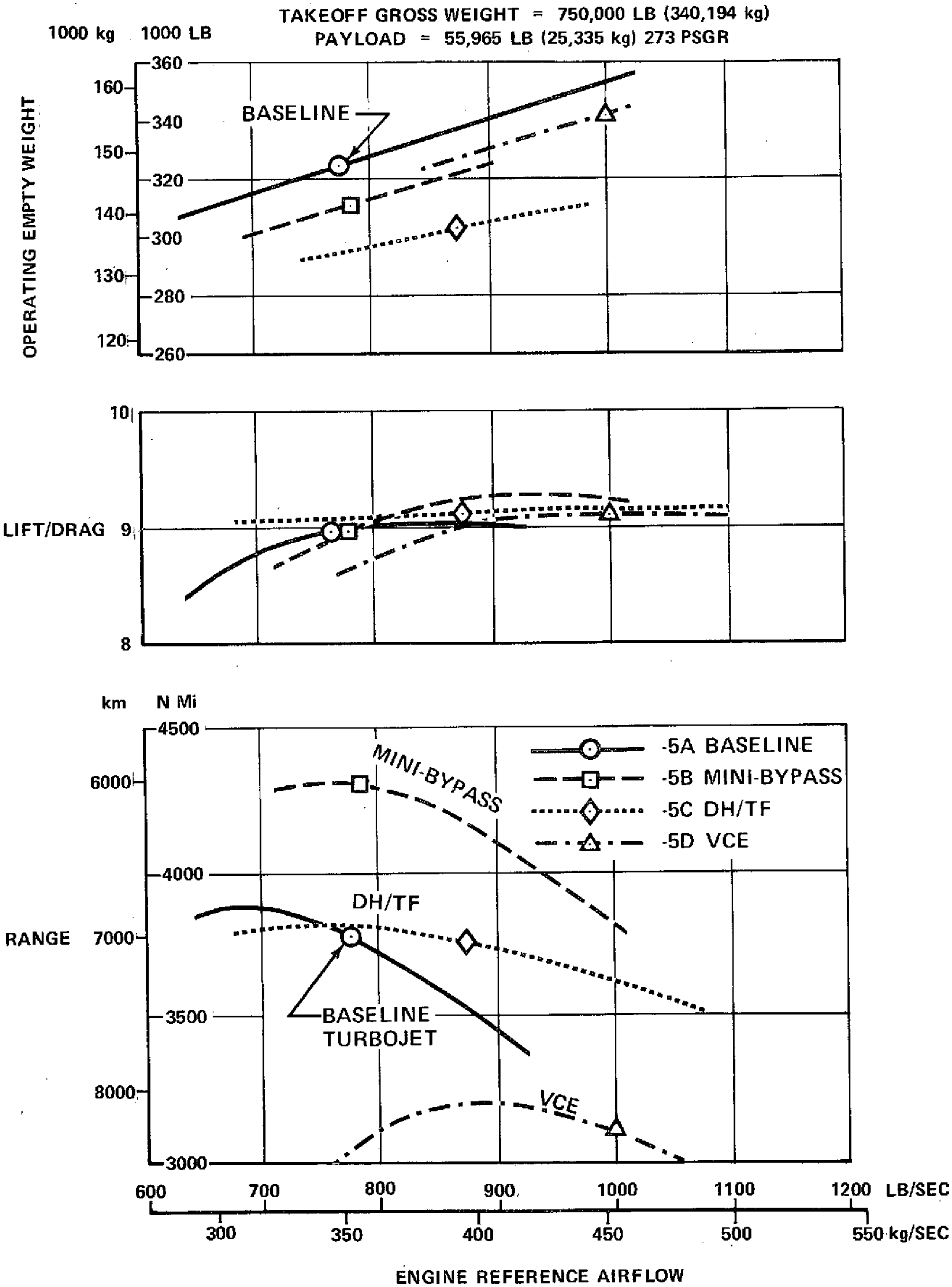


FIGURE A-3. ENGINE SIZING COMPARISON

is limiting. The VCE engine is sized at its maximum takeoff thrust, unthrottled, with no suppressor, and therefore cannot be sized smaller.

All engines studied in this report are stated by the engine manufacturers to be technically capable of design initiation in the 1978-80 time period. With a normal development period, any selected engine would not permit initial commercial operations until the late 1980's, which is probably later than desired for an advanced U.S. supersonic transport.

CONCLUSIONS

The following summarizes the significant conclusions from these studies:

1. The engine data supplied by the engine manufacturers as part of the recent NASA study contracts offer significant performance improvements for an advanced supersonic cruise aircraft than was available one year ago. More range and slightly reduced noise are now shown to be possible.
2. If one considers FAR Part 36 or FAR Part 36 minus two EPNdB as the AST noise requirements, then the mini-bypass turbojet cycle employing a plug nozzle/multi-chute suppressor exhaust system with a high level of jet noise suppression becomes the preferred AST engine based upon improved range performance. This selection is valid even considering degradation in effectiveness of this suppressor type, or by substitution of a DAC designed ejector nozzle/multi-chute suppressor/reverser exhaust system which is estimated to have improved performance but higher weight and drag.
3. The duct heating turbofan cycle at approximately FAR Part 36 or FAR 36 minus two EPNdB noise levels offers approximately the same range as the baseline dry turbojet airplane. It has the advantage of relying on lower levels of noise suppression and as such may offer improved potential for further noise reductions as technology improves. Also, it is less sensitive to performance degradation for missions with subsonic legs. This cycle warrants further evaluation.
4. The dual valve variable cycle engines all result in range losses as compared to the baseline airplane and appear to warrant no further evaluation.
5. Data received from the engine companies since engine selections were finalized for this study indicate that at least two new, improved variable cycle engines are now defined and ready for evaluation. Preliminary indications are that they are much improved over the dual valve VCE. The specific engines are the P&WA variable stream control engine and the GE double bypass dual cycle engine.

6. The figure-of-merit, range, as used in this study is satisfactory, however, other considerations are weighed carefully when airlines make engine selections. These include items such as timing, initial cost, operating cost, reliability, maintainability, experience, commonality, and safety. Such evaluations are beyond the scope of this study but need to be considered when applying these conclusions to a U.S. supersonic transport program.
7. Engine evaluations relative to suitability for mission performance must include detail installation design as detail design can have significant impact on the final result. Installation design, utilizing expertise unique to aircraft manufacturers, is required to insure the optimum, or best compromise, integrated propulsion system. Such items as propulsion control, cooling, integrated nozzle/reverser/suppressor, nacelle shape, and nacelle location must be addressed in close coordination with the engine companies. Only through analyses such as these can the engine be adequately evaluated as uninstalled data comparisons will not reveal the best configuration. Therefore, engine evaluations and eventual selections should include the mission performance of the integrated airframe/propulsion systems.

RECOMMENDATIONS

1. An improved version of the duct heating turbofan designated the variable stream control engine with significantly improved SFC for supersonic cruise should be evaluated on the baseline airplane.
2. A dual cycle, double-bypass concept with low jet exhaust noise for takeoff and low SFC for supersonic cruise should be evaluated in the baseline airplane.
3. The development testing of the DAC ejector/suppressor should be carried on in parallel with that for the plug nozzle/multi-chute suppressor in support of the mini-bypass engine evaluation.
4. A concurrent airplane evaluation study is recommended for advanced engine cycles as the engine concept design progresses. This will insure that realistic detail airplane installation design impacts will be accounted for in the evolving engine design.
5. The airframe manufacturer should contract directly with the major overseas carriers to evaluate credibility of engine selection parameters.

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AIRPLANE DEFINITION - BASELINE

The baseline configuration used for the present studies is a refined version of the final conceptual design resulting from the 1973 NASA technology studies (Report Douglas MDC J4394). Detailed analyses of that design revealed that the minimal wing outer panel thickness ratio, the location of maximum thickness, and the small depth available at the rear spar of the modified arrow wing produced a configuration which was not structurally efficient. Undesirably large wing tip twists and deflections were predicted and flutter speeds were less than required. A substantial weight penalty was necessary to alleviate the aeroelastic/flutter condition.

Trade studies were subsequently undertaken to alleviate these structural problems. An engine location study showed that moving the engines forward reduced the overhanging moment, increased the effective nacelle-pylon stiffness and reduced the overall weight of the wing-nacelle-pylon combination. Moving the inboard engine forward to the same chordwise station as the outboard had no significant effect on wave drag, but moving either or both engines forward of this location produced unacceptable values of wave drag. The optimum chordwise location of maximum thickness was evaluated since a rearward movement produced an increased depth at the rear spar. The wave drag penalties of various configurations were assessed while maintaining sufficient volume to contain the landing gear. The most efficient configuration selected holds the maximum thickness ratio constant at 2.25% between 60% and 75% of the chord at the root, constant at 3% between 40% and 65% of the chord at the trailing edge break (31% semi-span), and at 60% of the chord from the leading edge break (63.5% semi-span) to the tip. For these modifications, the maximum thickness ratio of the outer panel (63.5% semi-span to tip) was then varied from 2% to 4% and overall efficiency evaluated by minimizing the sum of the wing structural weight and the drag equivalent weight as shown in Figure 1-1. Based on these results, an outer panel thickness ratio of 3% was selected. The structural and aerodynamic parameters were analytically derived using the Structural Optimization FORMAT subroutine program and the wave drag option of the Arbitrary Body program. Flutter analysis was included and no fuel was placed in the outboard wing panels. The L/D decrement for optimizing the design while solving the structural problems was 0.43.

From the weights standpoint the baseline configuration (-5A) is a derivative of Model D3230-2.2-4 (-4). The -4 configuration was evaluated in depth during the 1973 NASA technology studies. Differences between the -4 and -5A consist of nominal increases in gross weight and wing area, along with minor changes in wing, fuselage and vertical tail geometry. Engine packaging and locations have also been changed. However, the similarities are such that much of the weight information generated for the -4 is adaptable to the -5A configuration.

Weights for both the -4 and -5A are based on methodologies developed from previous commercial and military programs. Primary among these is a computerized mass properties estimation system designed MAPES. This system utilizes 300 inputs, consisting of loads, criteria, geometry, etc., to generate a 400 component structure and systems weight breakdown in a MIL-STD-1374 format. The MAPES system was initially developed to evaluate subsonic transport aircraft, and at this time, approximately 63 percent of the program output is valid for supersonic aircraft. The exceptions are the wing box structure and nacelle inlet.

The weight of the -5A box structure is estimated using a multi-station analysis methodology developed for low aspect ratio wings, utilizing finite element

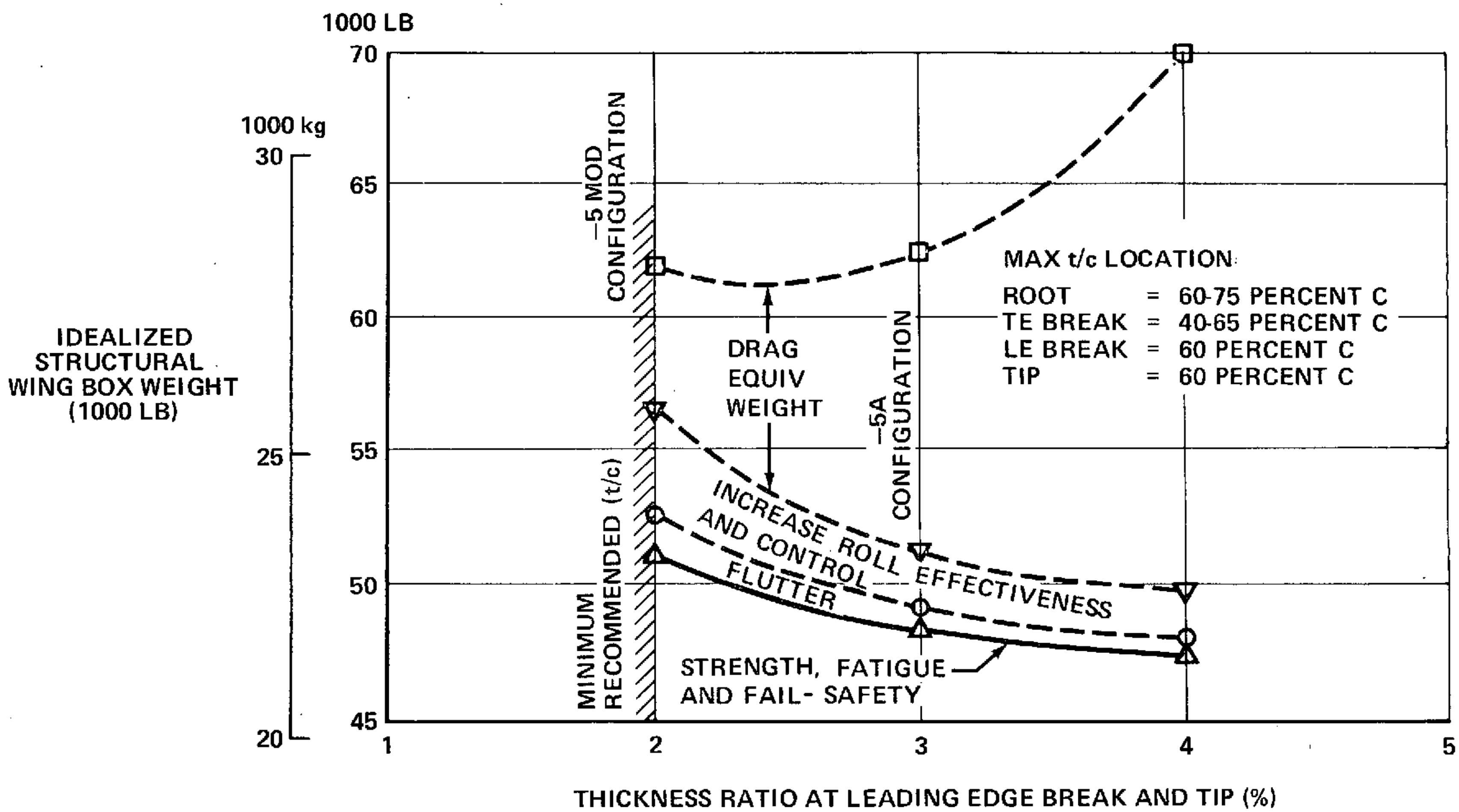

FIGURE 1-1. AST WING STRUCTURAL BOX OPTIMIZATION

analyses verified by structural design layouts. Details of the finite element analyses are discussed in the Structural Analysis paragraph of Section 2.

Inlet weights are developed from inhouse design studies. Variations from point designs are effected using weight relationships, presented in Technical Report SEG-TR-67-1[1], as a guide. Engine and exhaust system weights for all configurations are based on engine manufacturer supplied weight data. The weight summary for the baseline configuration is shown in Table 1-1. Also included are the increments for aeroelasticity and flutter.

The three-view for the baseline model D3230-2.2-5A (hereafter referred to as the -5A) is shown in Figure 1-2. This constitutes a realistic base from which individual engine variations could be readily evaluated during the course of this study.

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(1) E. L. Crosthwait, I. G. Kennon, Jr. and H. L. Roland, Preliminary Design Methodology for Air Induction Systems, Technical Report, SEG-TR-67-1, January, 1967

TABLE 1-1
BASELINE CONFIGURATION (–5A) WEIGHT SUMMARY

ITEM	ENGLISH UNITS	METRIC UNITS
WEIGHTS:	LB	kg
WING	75,347	34,177
H-TAIL	3,960	1,796
V-TAIL	3,807	1,727
FUSELAGE	47,713	21,642
LANDING GEAR	36,792	16,689
FLIGHT CONTROLS	9,115	4,134
NACELLE/INLET	14,730	6,681
PROPULSION (LESS FUEL SYSTEM)	70,190	31,838
FUEL SYSTEM	3,820	1,733
EMERGENCY POWER UNIT	950	431
INSTRUMENTS	1,227	557
HYDRAULICS	5,684	2,578
PNEUMATICS	1,332	604
ELECTRICAL	4,850	2,200
NAVIGATION AND COMMUNICATION SYSTEM	2,756	1,250
FURNISHINGS	24,478	11,103
AIR CONDITIONING	4,854	2,202
ICE PROTECTION	489	222
HANDLING PROVISIONS	90	41
PENALTY FLUTTER AND AEROELASTICITY	2,860	1,297
STRUCTURAL WEIGHT INCREMENT	–	–
MANUFACTURERS EMPTY WEIGHT	315,044	142,902
OPERATIONAL ITEMS	8,096	3,672
OPERATIONAL EMPTY WEIGHT	323,140	146,574

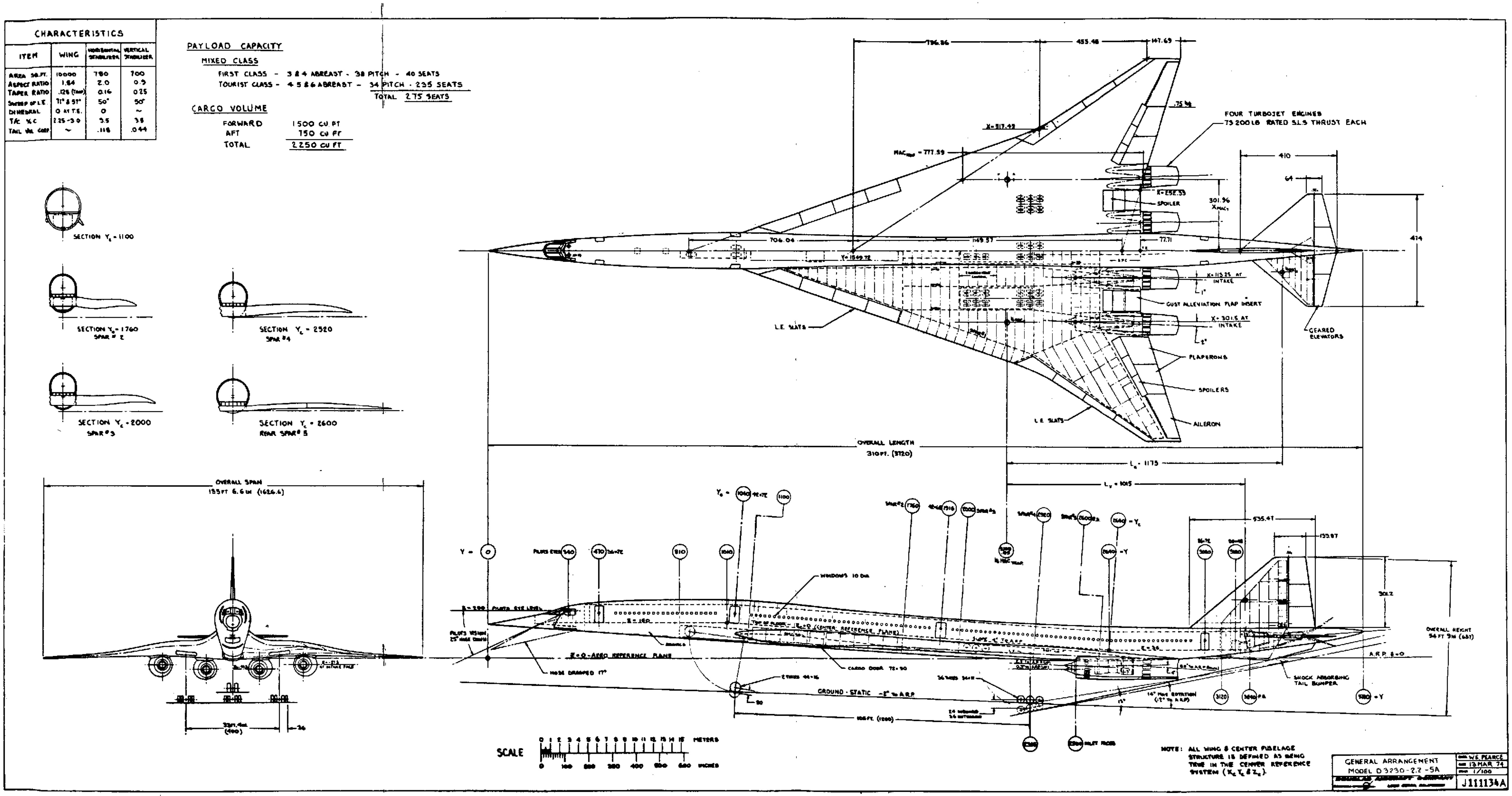

FIGURE 1-2. AST BASELINE CONFIGURATION (-5A)

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ENGINE DEFINITION - BASELINE

The engine defined for the baseline airplane is based on data obtained from the engine manufacturers during the 1973 NASA engine contract studies. From a screening of these data for the dry turbojet, afterburner turbojet, duct heating turbofan, and several valved variable cycle engines it had been concluded that the dry turbojet incorporating 1975 technology was the most promising from the standpoint of providing a minimum operating cost design. A conceptual turbojet engine was, therefore, sized to meet FAR Part 36 noise constraints and an in-house engine deck was derived and utilized in the determination of airplane performance.

The engine size is 773 lb/sec (351 kg/sec) inlet corrected airflow at maximum combustor exit temperature at sea level Std. + 18°F (10°C) day static takeoff operation. The design cycle characteristics and ratings are shown in Table 1-2. The engine exhaust system is a convergent-divergent ejector nozzle, incorporating a suppressor stowed in the nozzle within the ejector shroud and utilizing trailing edge buckets as the exit area control and reverser. A sketch of the engine is shown in Figure 1-3. The installed engine is shown in Figures 1-4 and 1-5.

Engine weights, dimensions, scaling equations and cost data are presented in Table 1-2. Cost data are based on P&WA cost information provided as part of their Advanced Supersonic Propulsion System Technology studies conducted under contract to NASA Lewis in 1973. Costs have been escalated to 1973 by DAC based on 1972 dollar values provided in the engine manufacturers' study.

TABLE 1-2

BASELINE TURBOJET
ENGINE CHARACTERISTICS SUMMARY
773 LB/SEC (351 kg/SEC) RATED AIRFLOW

DESIGN CYCLE CHARACTERISTICS

CYCLE PRESSURE RATIO 18:1

COMBUSTOR EXIT TEMP (T.O.) 2600°F (1700°K)
(MAX CLIMB) 2500°F (1644°K)
(MAX CRUISE) 2400°F (1589°K)

TAKEOFF RATINGS [STD DAY + 18°F (10°C)]

MAXIMUM THRUST (SLS) – LB 73,173
(kN) (325.49)

MAXIMUM THRUST (SL, 0.3M, UNINSTALLED) – LB 66,637
(kN) (296.41)

THRUST AT 1500°F EGT (SL, 0.3M UNINSTALLED) – LB 58,585
(kN) (260.60)

WEIGHT

ENGINE – LB 12,902
(kg) (5852.3)

ENGINE + NOZZLE/REVERSER/SUPPRESSOR – LB 16,982
(kg) (7678.5)

DIMENSIONS

ENGINE INLET GAS
FLOW PATH DIAMETER – IN. 66.4
(m) (1.687)

ENGINE MAX DIAMETER – IN. 97.0
(m) (2.464)

HUB-TO-TIP RATIO
(AT PLANE OF ATTACH FLANGES) 0.42

LENGTH – INLET
FLANGE TO EXHAUST PLANE – IN. 358.4
(m) (9.103)

SCALING FACTORS

WEIGHT $\frac{WT}{WT_O} = \left(\frac{WAT2}{773}\right)^{1.16}$

DIAMETER $\frac{D}{D_O} = \left(\frac{WAT2}{773}\right)^{0.5}$

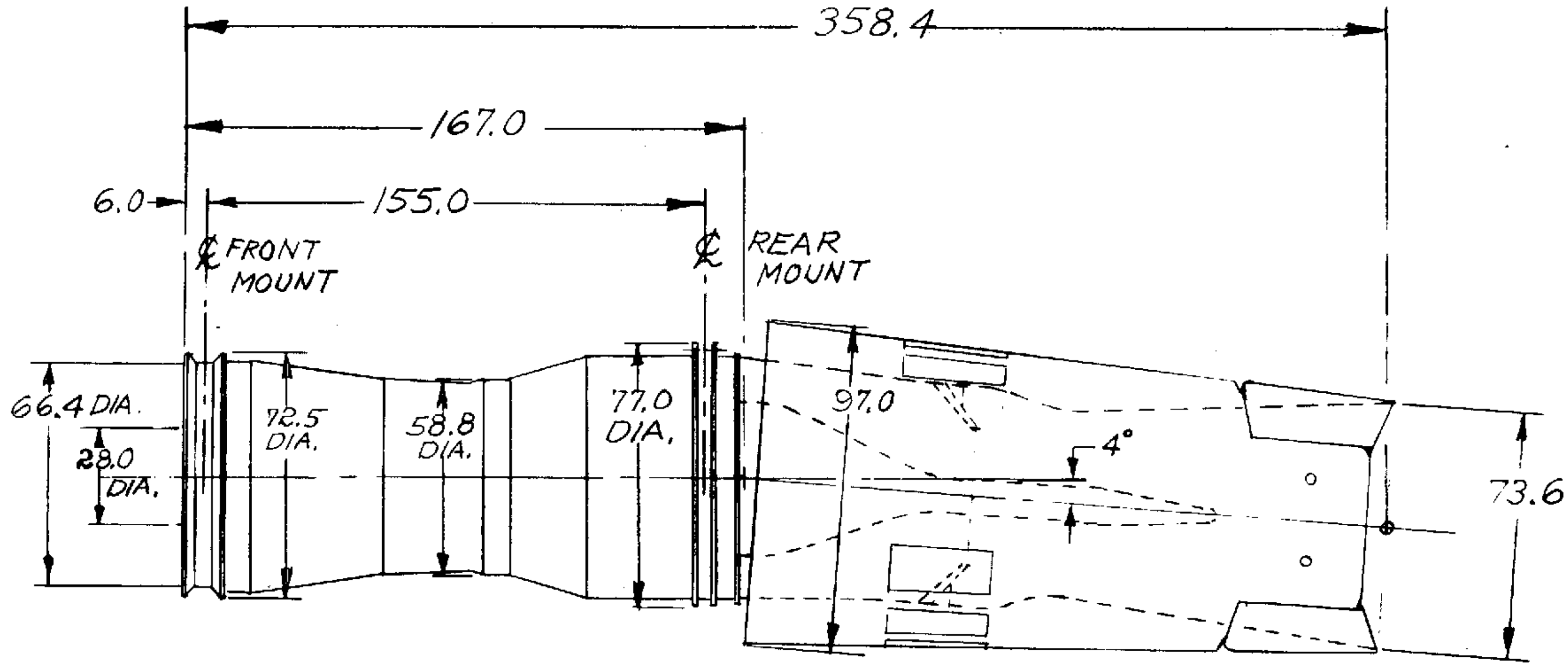
LENGTH $\frac{L}{L_O} = \left(\frac{WAT2}{773}\right)^{0.43}$

COST*

WITHOUT SUPPRESSOR $2.73M
WITH SUPPRESSOR $3.02M

SCALING FACTOR $\frac{COST}{COST_O} = \left(\frac{WAT2}{773}\right)^{0.64}$

*BASED ON
- 1973 DOLLARS
- 1975 ENGINE TECHNOLOGY
- 3000 ENGINE PRODUCTION RUN
- PRICES INCLUDE ALL DEVELOPMENT COSTS PLUS FIVE-YEAR PRODUCT SUPPORT AFTER CERTIFICATION BASED ON ONE ENGINE MODEL



AIRFLOW 773 LB/SEC (350.6 kg/SEC)

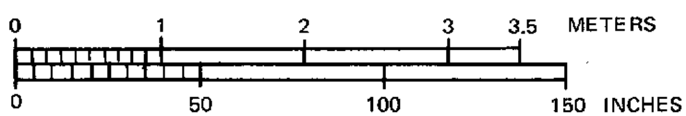


FIGURE 1-3. BASELINE TURBOJET ENGINE

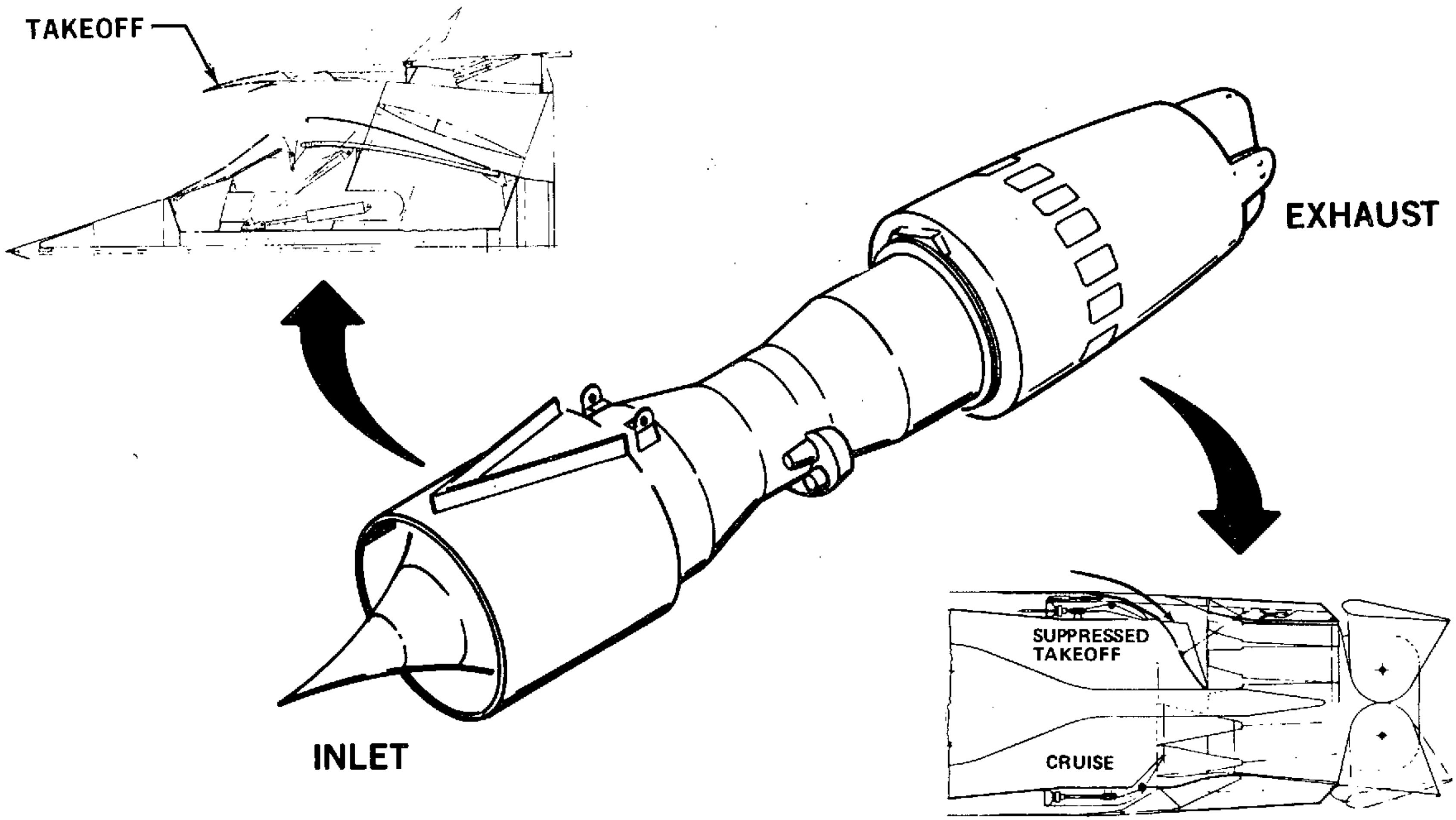
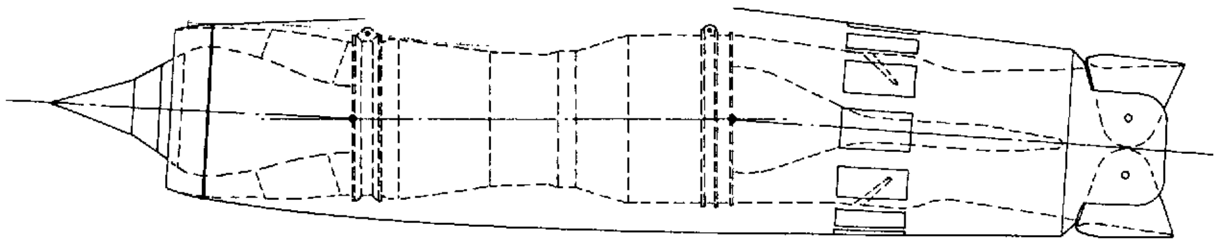


FIGURE 1-4. BASELINE TURBOJET POWER PLANT INSTALLATION DESIGN



73,173 LB RATED SLS THRUST (325.49 (kN)

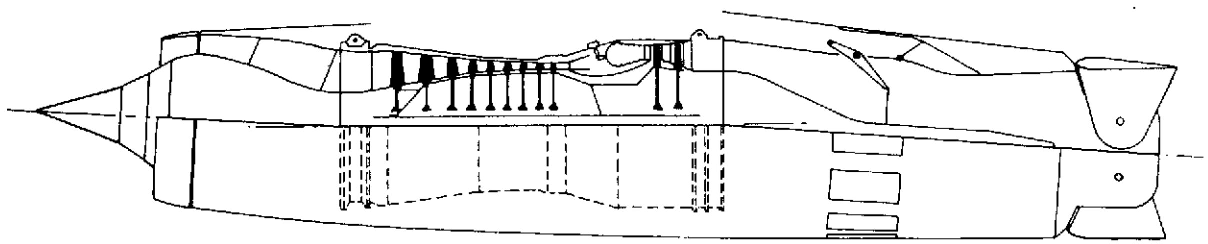


FIGURE 1-5. BASELINE TURBOJET ENGINE INSTALLATION

PROPULSION SYSTEM PERFORMANCE - BASELINE

Uninstalled Performance

The uninstalled engine performance includes the effects of:

° U.S. 1962 model atmosphere	
° Inlet recovery	Figure 1-6
° DAC internal nozzle velocity coefficient	
° Customer compressor air bleed	0.28% of engine airflow
° Customer power extraction	200 HP (149 kW)
° Jet A Fuel, Lower Heating Value	18,400 BTU/lb. (4.34×10^7 J/kg)
° No losses for acoustical treatment	

Installed Performance Analysis

The analysis of the propulsion system performance includes the determination of the inlet recovery and drag characteristics, and an estimation of nacelle drag characteristics which are combined with the uninstalled engine performance to produce the installed propulsion system performance.

The inlet performance and the nacelle analysis include an evaluation of the following items:

- Inlet spillage drag
- Inlet bypass drag
- Engine and ECS cooling airflow drag
- Nacelle afterbody drag
- Nacelle wave drag

These characteristics are estimated based on both theoretical and empirical methods.

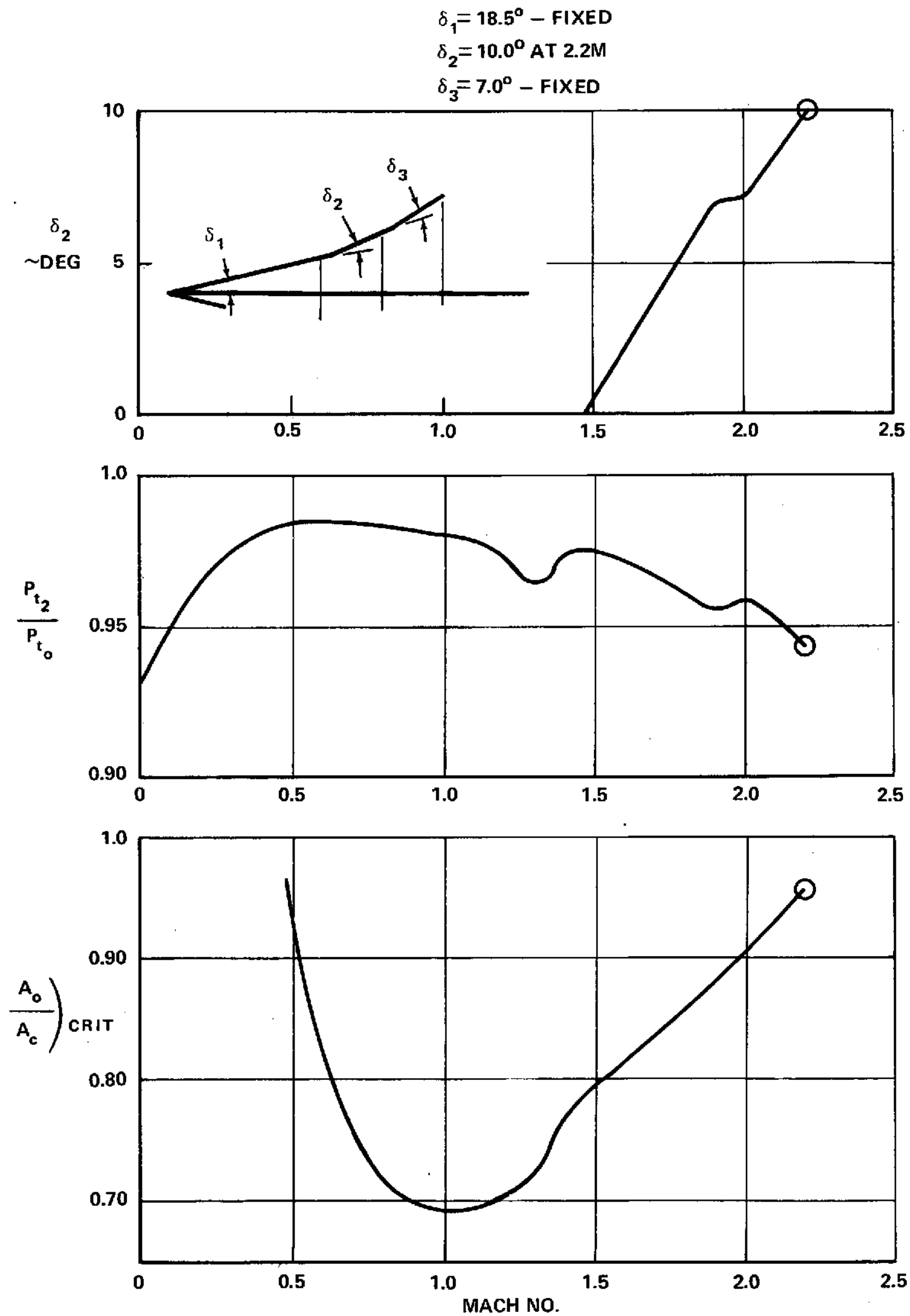


FIGURE 1-6. INLET PERFORMANCE

An inviscid analysis is used to define an inlet cone angle schedule to avoid shock ingestion and the attendant high inlet total pressure distortion levels. The same analysis provides the inviscid total pressure losses, the additive drag component of inlet spillage drag, and the mass-flow ratio for the inlet operating at critical conditions. The theoretical results are combined with empirical correlations to define the inlet total pressure recovery variation shown in Figure 1-6. Also shown in the figure is the variation of inlet critical mass-flow ratio and the inlet cone angle schedule. It should be noted that the first cone has been assumed to be fixed. Shown in Figure 1-7 is the mass-flow ratio for the inlet boundary layer bleed airflow. This schedule has been derived from a correlation of inlet test results.

The engine airflow schedule for the baseline turbojet engine is shown in Figure 1-8. The installed inlet performance for this engine is shown in Figure 1-9. As shown by the upper graph in the figure, the inlet airflow supply provides an adequate match with the engine airflow demand. The inlet is sized at the design point of 2.2M. The sized capture area is 23.8 ft.^{2} (2.21m^2). The engine and ECS cooling airflow are based on an allowance of 2 percent of inlet capture area airflow for the environmental control system (ECS) cooling and engine compartment ventilation and nozzle cooling.

The nacelle drag coefficient buildup is shown in the lower graph in Figure 1-9. The inlet drag characteristics are calculated by combining the mass-flow ratio characteristics with empirical drag coefficient correlations. For the convenience of engine sizing studies, the nacelle skin friction drag is included in the installed engine performance. The skin friction coefficients are based on fully turbulent flat plate adiabatic wall boundary layer data with transition at the leading edge and the resulting drag is shown in the figure. Since

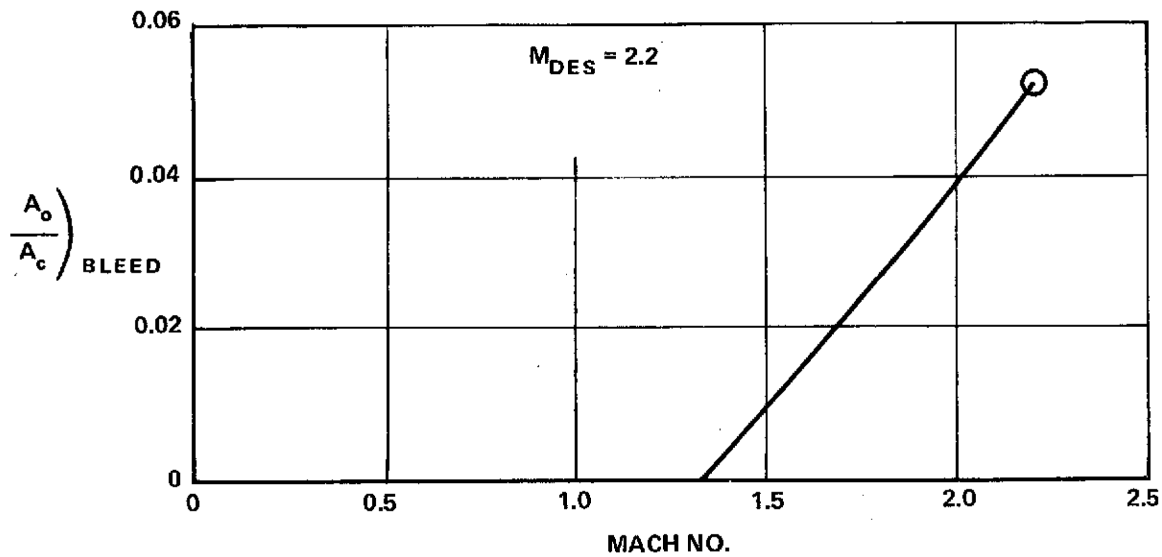


FIGURE 1-7. ESTIMATED BLEED MASS-FLOW-RATIO

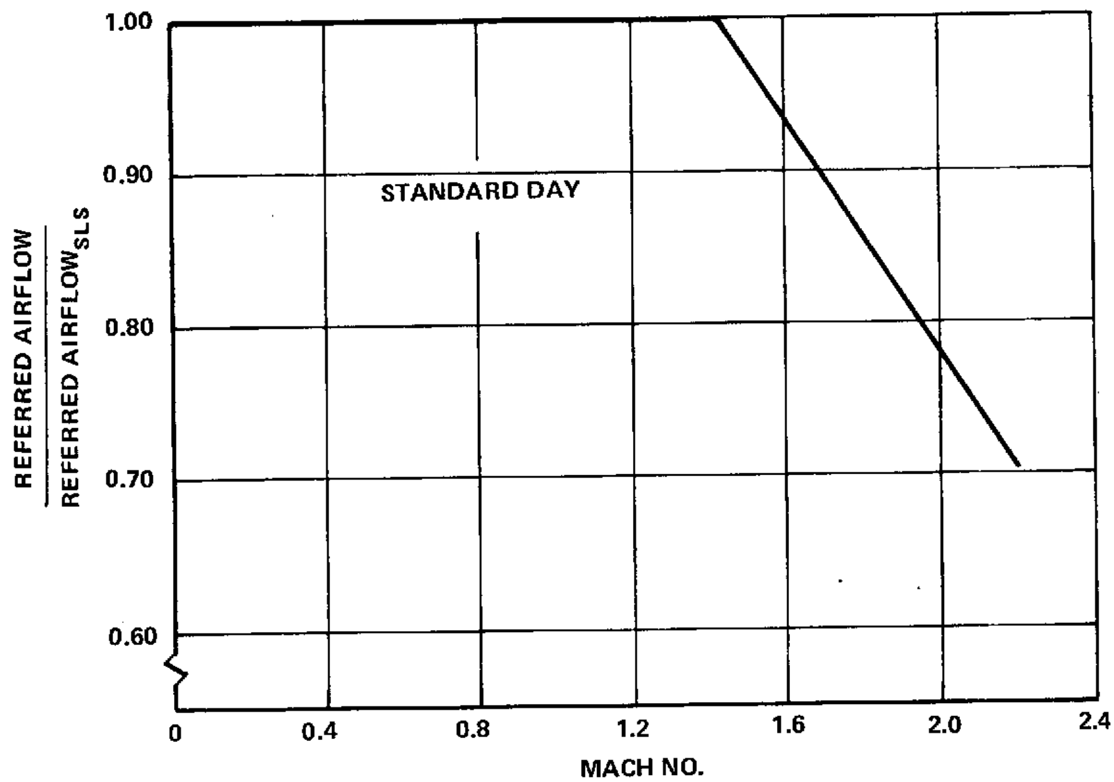


FIGURE 1-8. ENGINE AIRFLOW SCHEDULE

DAC TURBOJET

$A_C = 23.8\ FT^2\ (2.21 m^2)$

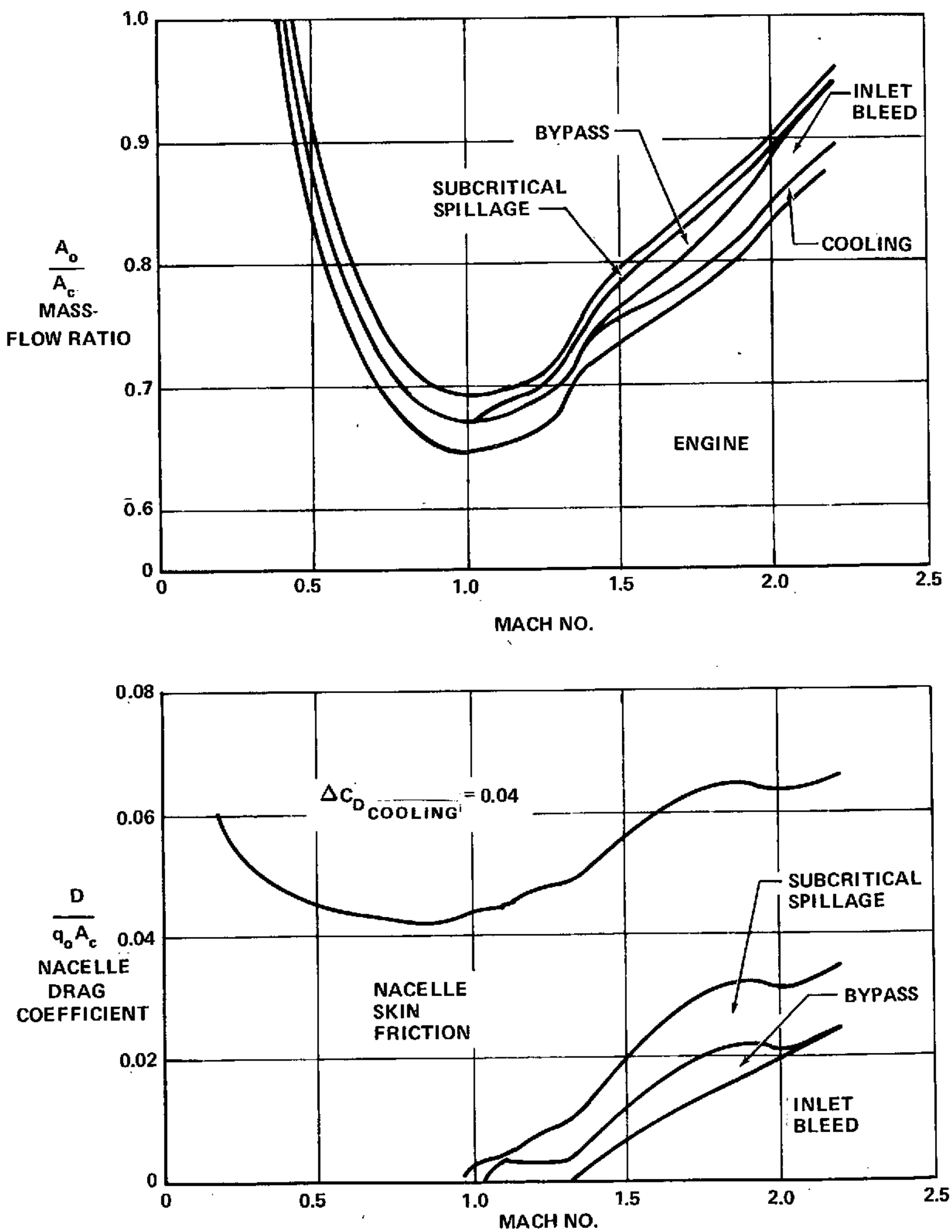


FIGURE 1-9. INSTALLED INLET PERFORMANCE

the thrust recovery of the cooling airflow is included in the nozzle thrust, the cooling drag is equal to the full ram drag of the cooling airflow. As indicated on the figure, the cooling drag coefficient is constant at 0.04.

The nacelle afterbody drag is dependent on the nozzle exit area and flight Mach number. The maximum nozzle area is sized at 2.2M, maximum climb thrust. The engine dependent boattail drag at this condition is zero. As nozzle area decreases for lower Mach numbers and reduced power settings, the boattail drag increases. The variations in drag coefficient relative to the design condition along the aircraft climb path at maximum climb thrust and for subsonic flight are shown in Figures 1-10 and 1-11, respectively.

The nacelle wave drag in the presence of the aircraft, including the supercritical spillage drag and the design afterbody drag is treated as part of the aircraft wave drag.

Performance Results

Installed propulsion system performance is generated by correcting the DAC turbojet uninstalled engine performance data for the installation effects described above.

The climb performance characteristics are generated along the aircraft flight path shown in Figure 1-12. Uninstalled and installed thrust for the takeoff power setting (EGT limited for noise) are shown in Figure 1-13. Figures 1-14 and 1-15 present the uninstalled and installed referred thrust and SFC, respectively, for maximum climb thrust along the climb flight path. Uninstalled and installed supersonic cruise, subsonic cruise (for alternate mission), and hold performance are shown in Figure 1-16 through 1-18. Figure 1-19 presents the installed characteristics used along the descent flight path.

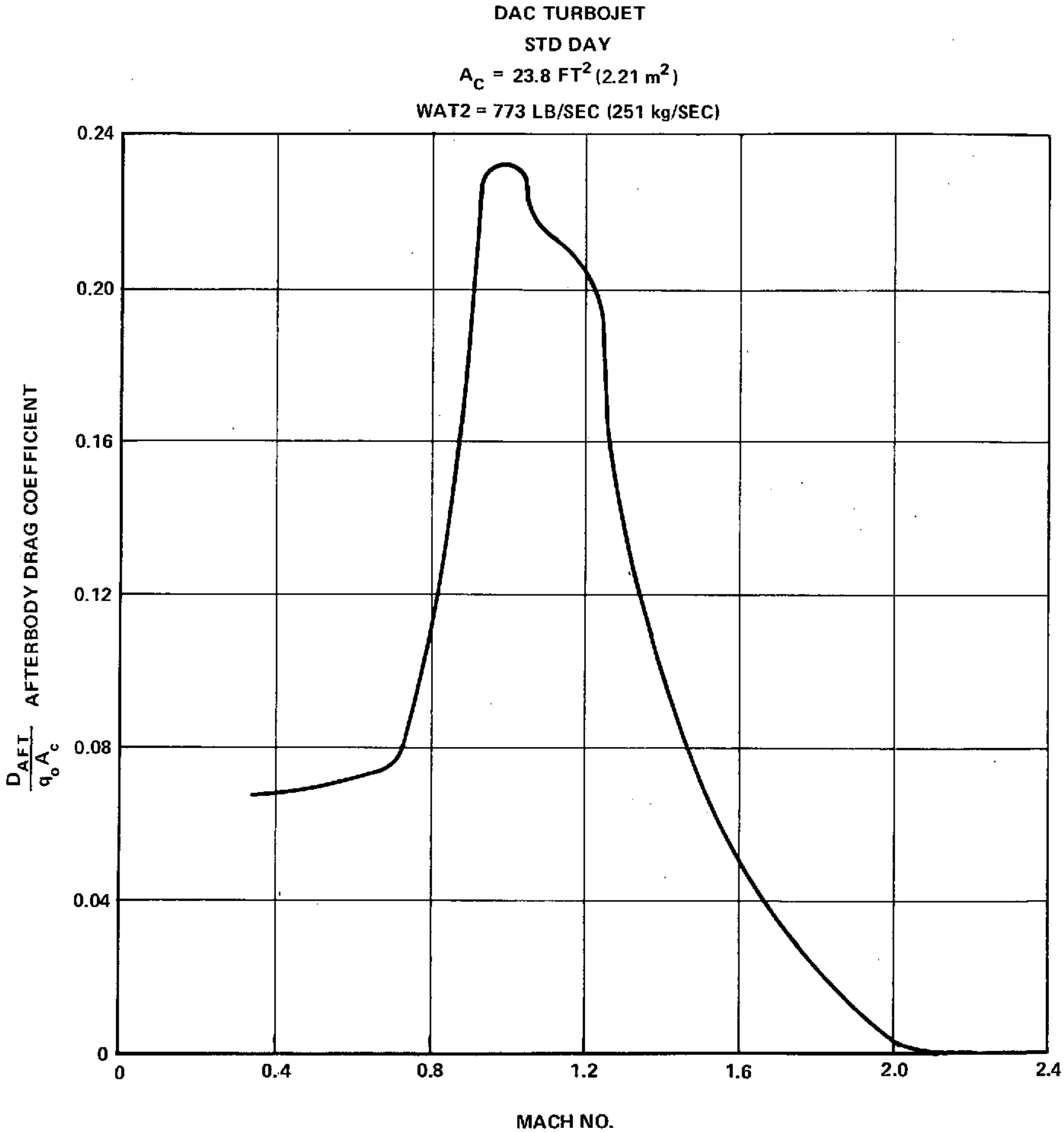


FIGURE 1-10. CLIMB AFTERBODY DRAG

DAC TURBOJET

STD DAY

A_C = 23.8 FT^2 (2.21 m^2)

WAT2 = 773 LB/SEC (351 kg/SEC)

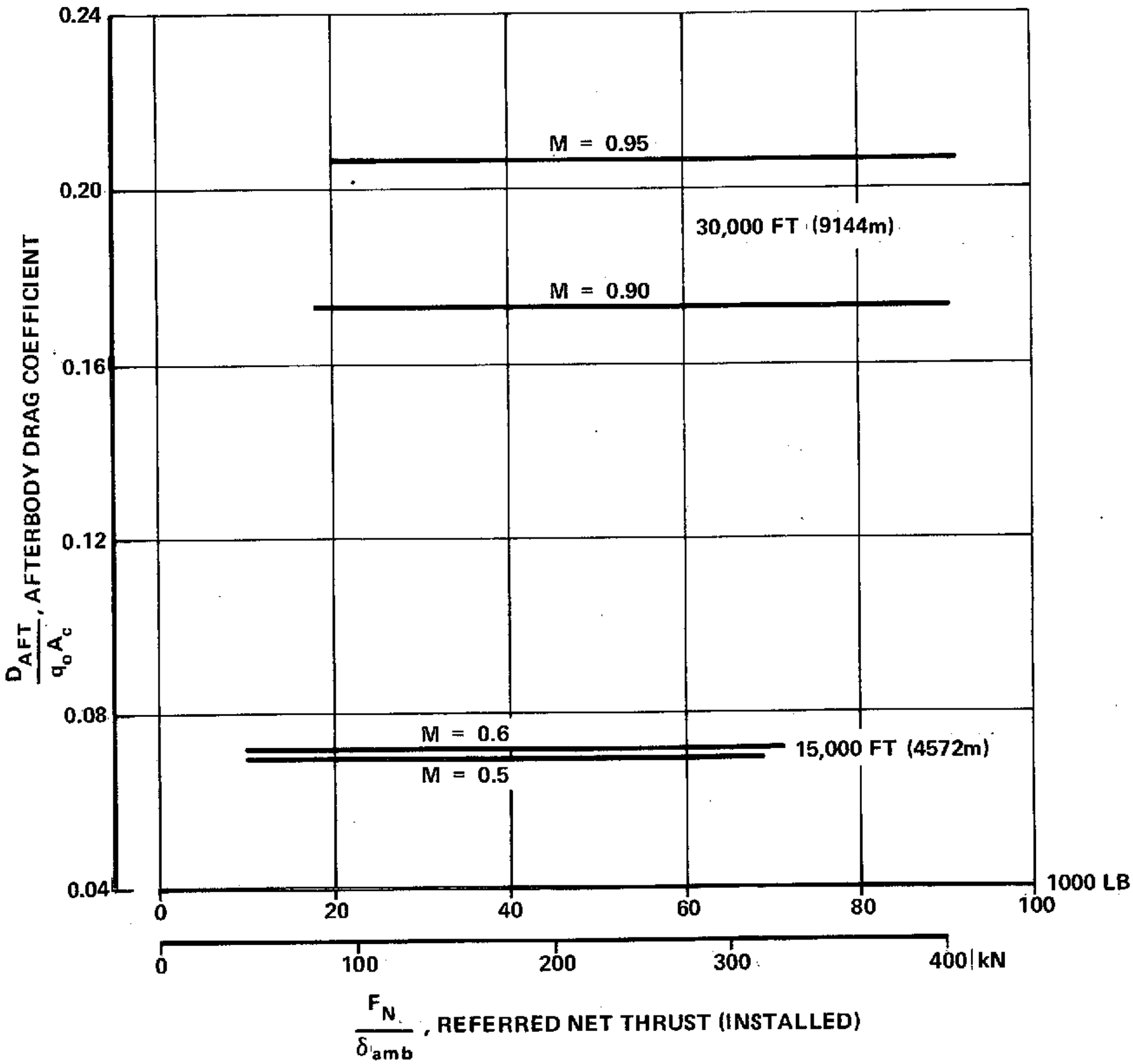


FIGURE 1-11. SUBSONIC AFTERBODY DRAG

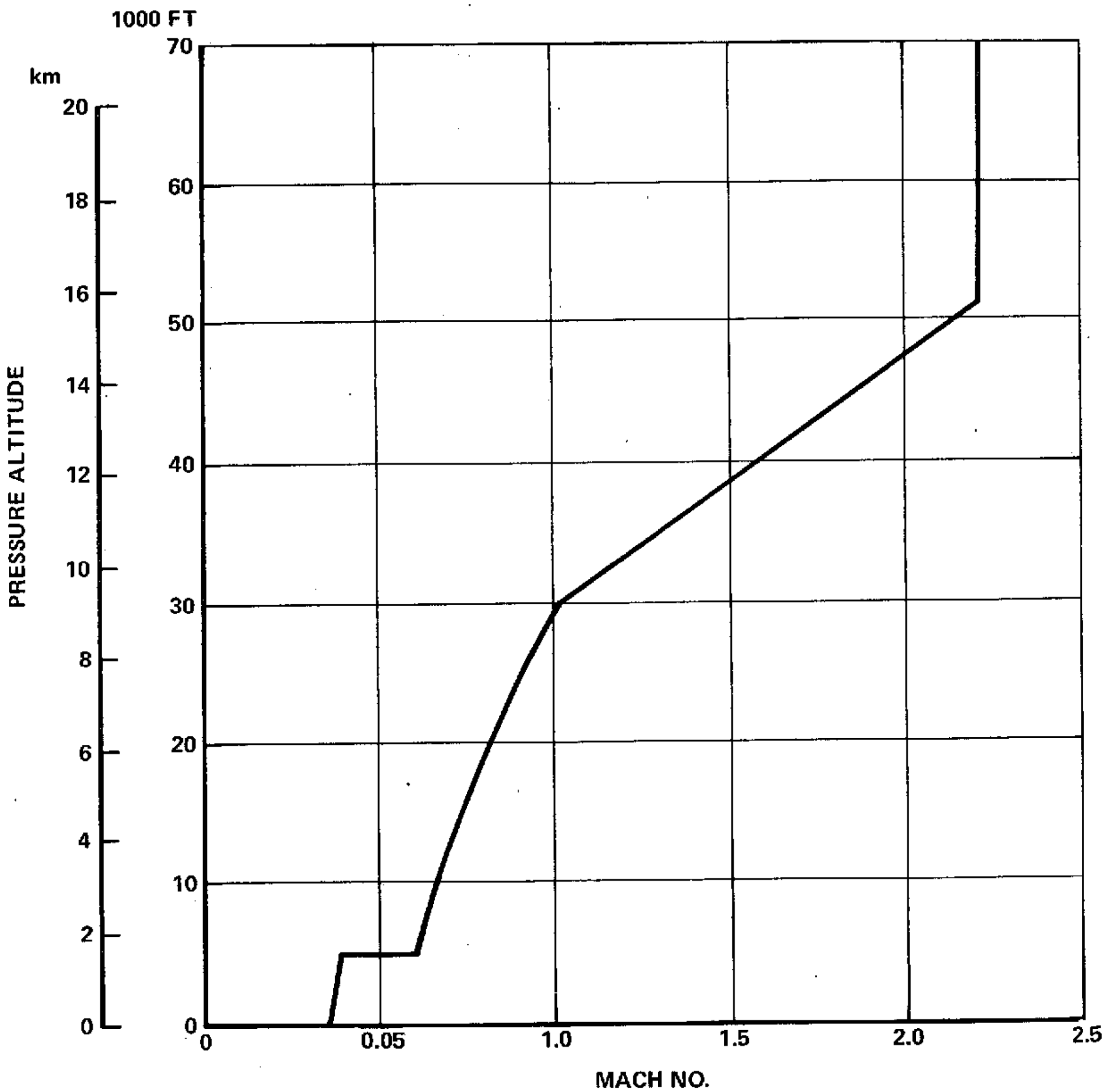


FIGURE 1-12. CLIMB FLIGHT PATH

DAC TURBOJET
SEA LEVEL, STD DAY
WAT2 = 773 LB/SEC (351 kg/SEC)
SLS RATING = 73,173 LB (325.49 kN)
100% SLS THRUST (UNINSTALLED) = 66,550 LB (296.03 kN)

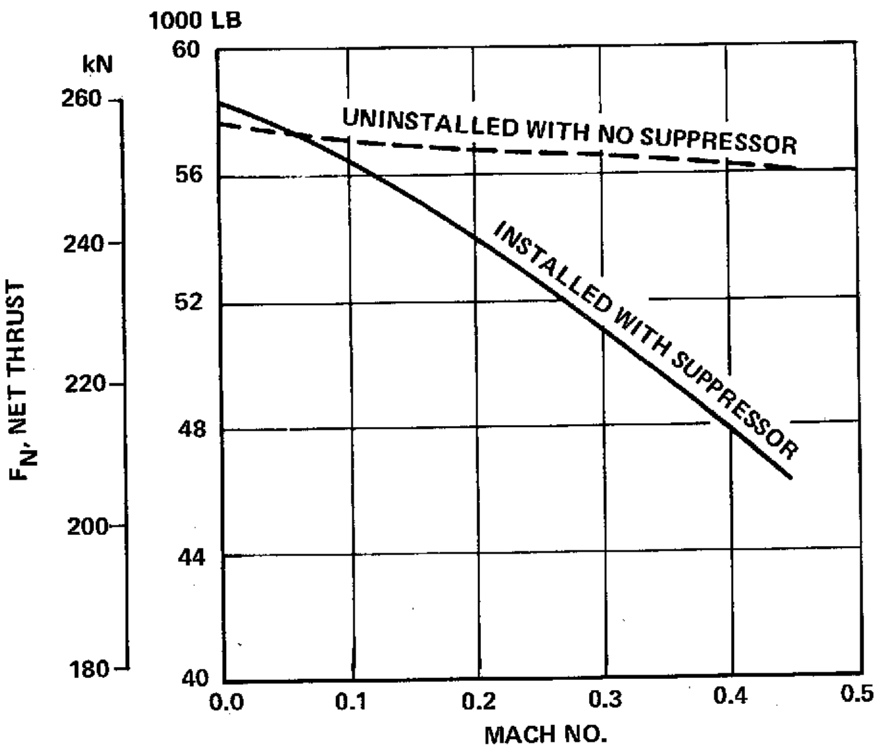


FIGURE 1-13. TAKEOFF PERFORMANCE

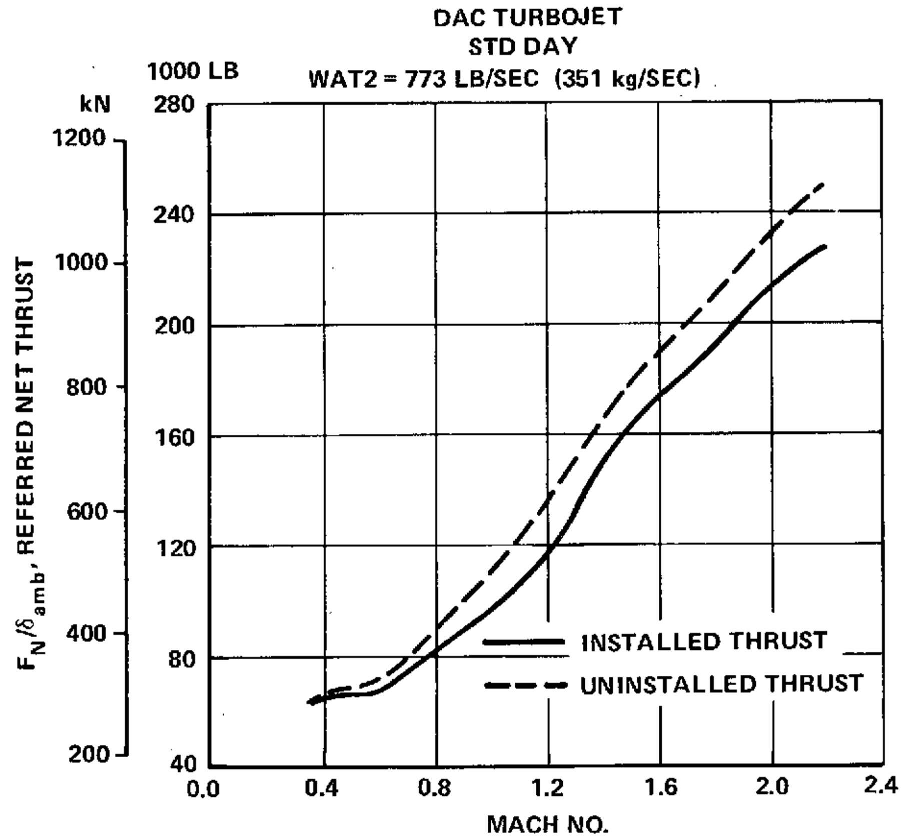


FIGURE 1-14. CLIMB THRUST

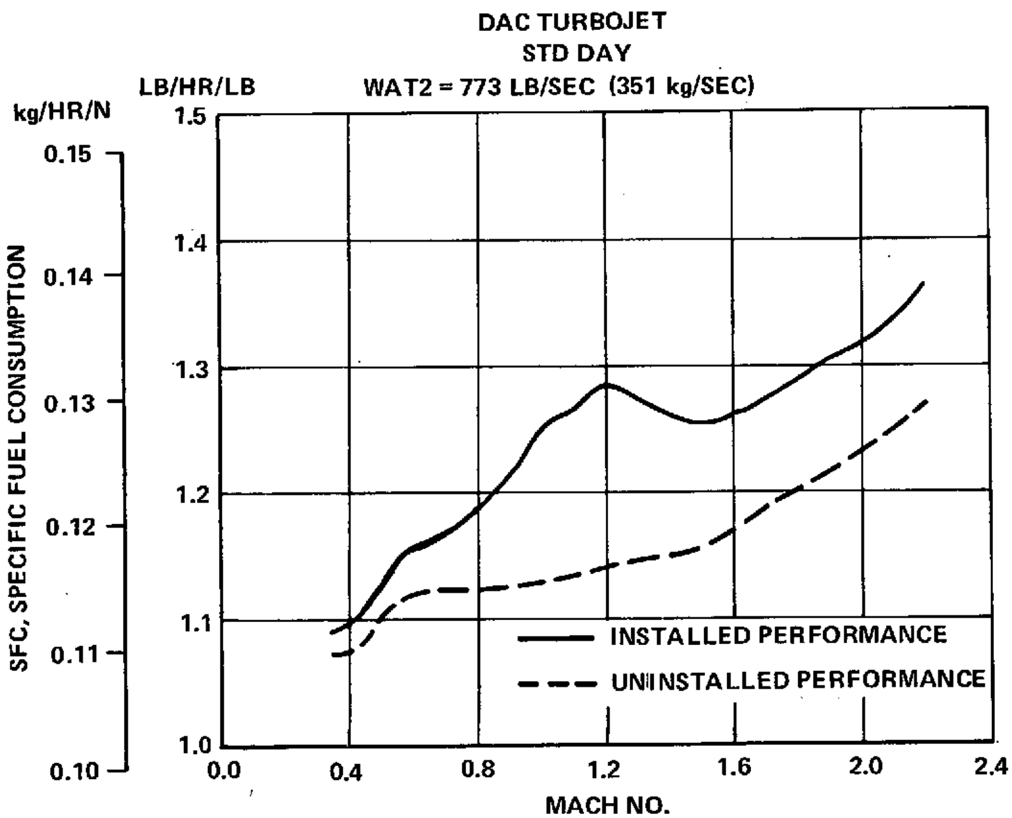


FIGURE 1-15. CLIMB SFC

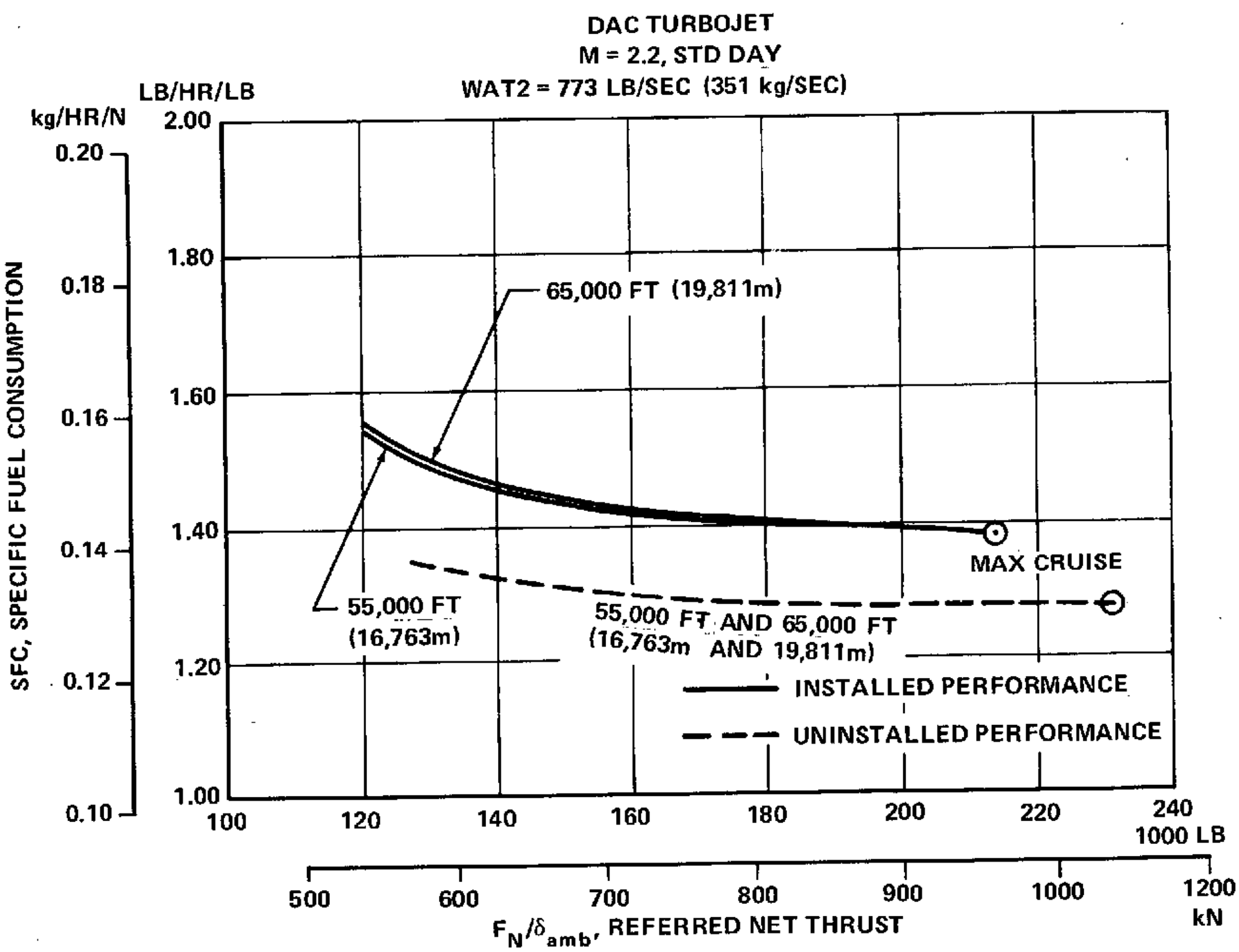


FIGURE I-16. SUPERSONIC CRUISE PERFORMANCE

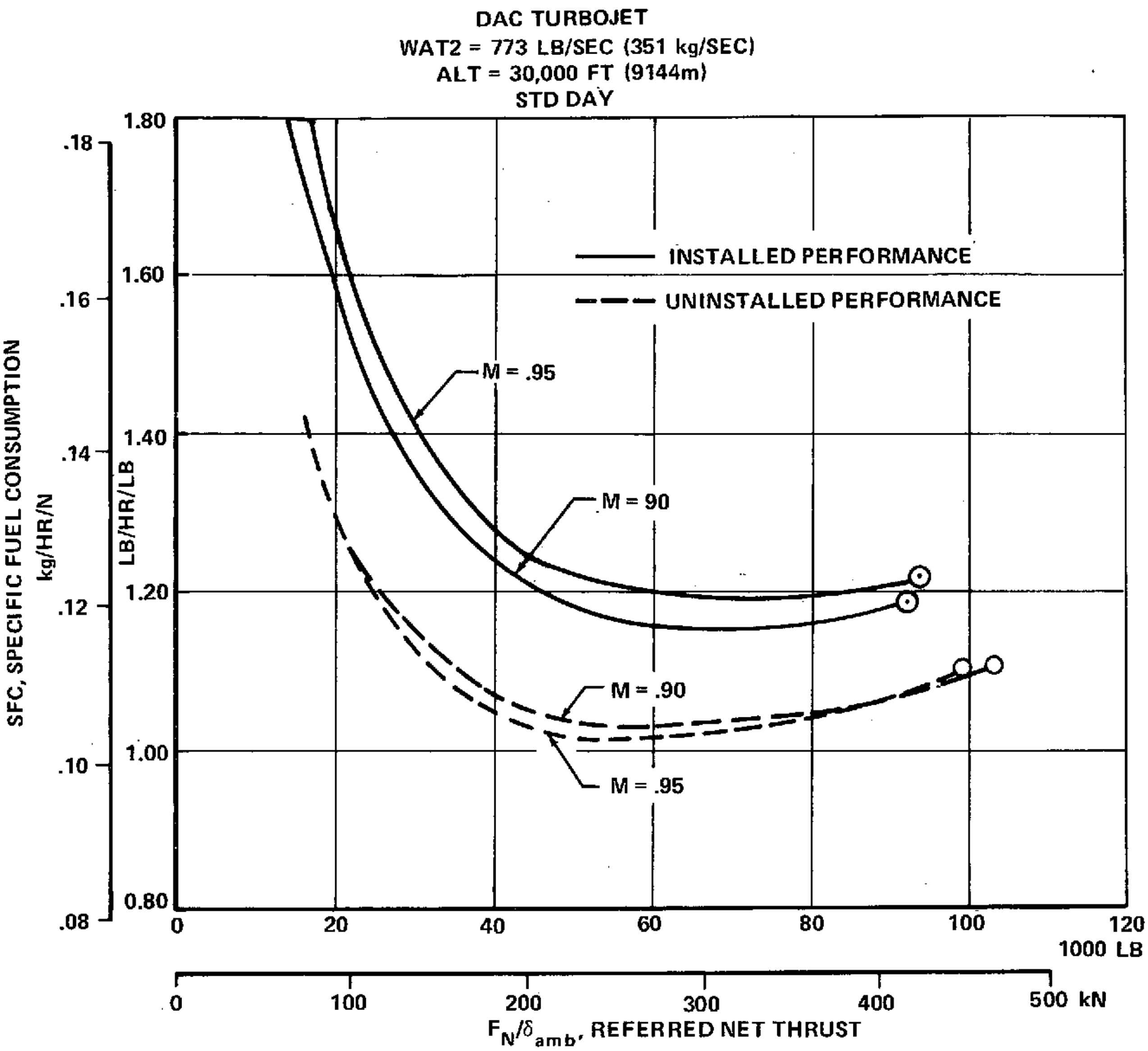


FIGURE 1-17. SUBSONIC CRUISE PERFORMANCE

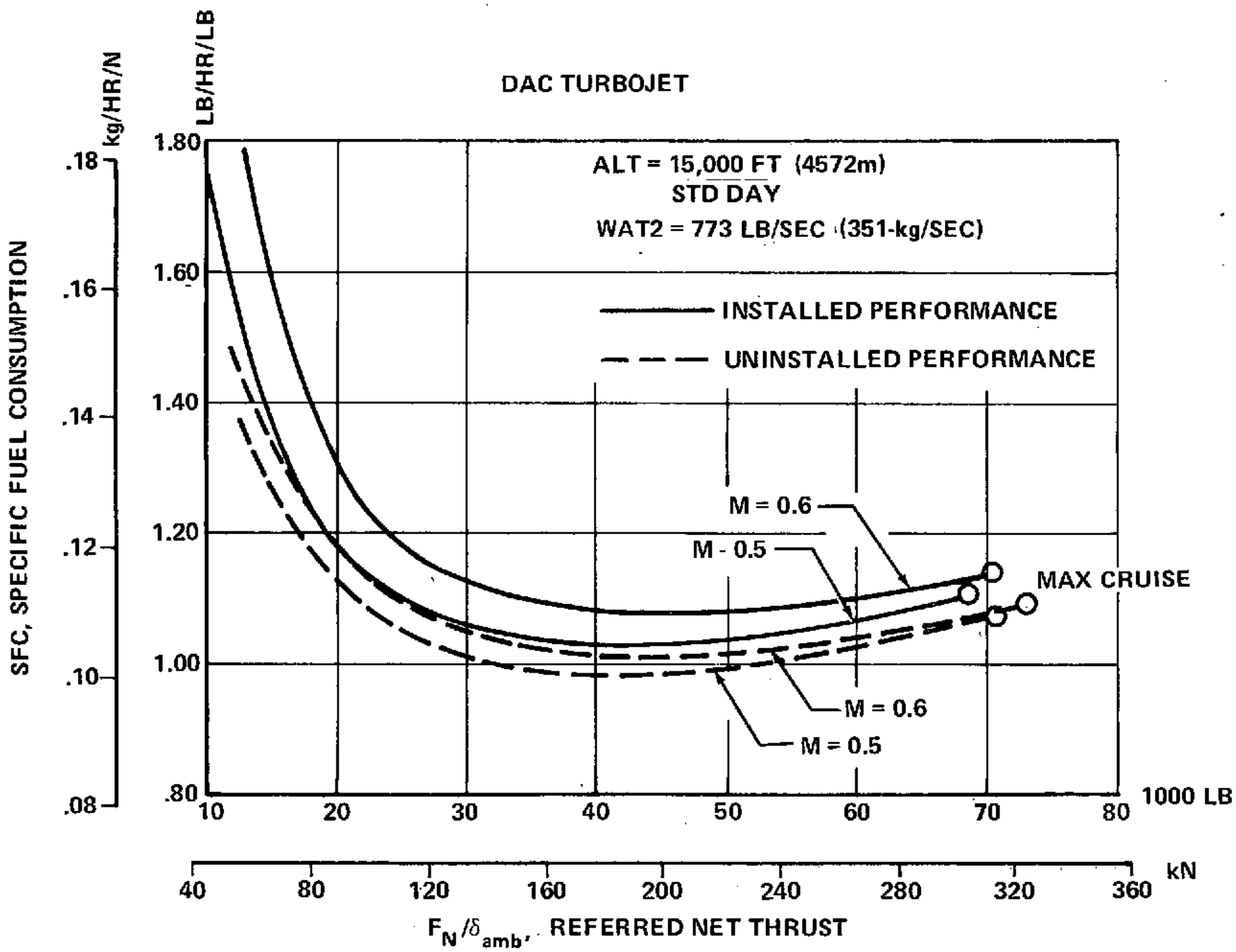


FIGURE 1-18. LOITER PERFORMANCE

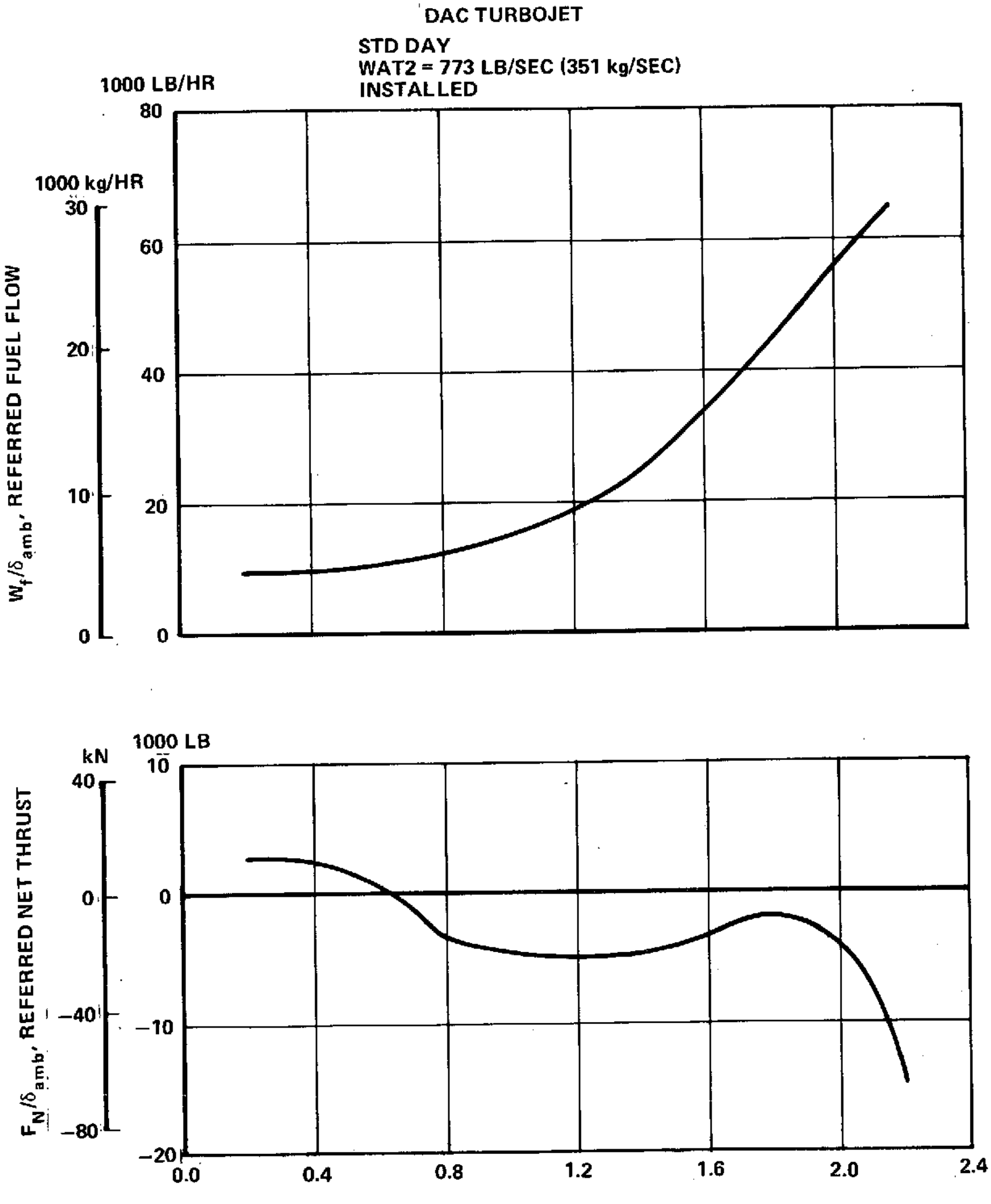


FIGURE 1-19. IDLE PERFORMANCE

Characteristics are tabulated below for the -5A baseline configuration, including the penalties for flutter/aeroelastic/drag optimization [220 n. miles (408 km) for L/D decrement, 70 n. miles (130 km) for the increase in weight] and basic airplane and engine data updating subsequent to the data included in the technology study report (MDC J4394). The mission and reserve ground rules are shown in Figure 1-20.

Takeoff Gross Weight	750,000 lb (340,194 kg)
Payload (273 Passengers)	55,965 lb (25,385 kg)
Takeoff Field Length	10,700 ft (3261 m)
Height at 3.5 n.mi. (6.5 km) Takeoff point	1,256 ft (383 m)
Sideline Noise Level (2270 ft.) (692m)	104.4 EPNdB
Takeoff Noise Level (cutback at 3.5 n.mi.) (6.5 km)	105.3 EPNdB
Approach Noise Level (370 ft. alt) (122 m)	107 EPNdB
Range	3,782 n.mi. (7006 km)
Initial Cruise Altitude	55,300 ft (16.9 km)
Direct Operating Cost (1973 $)	1.84 cents/seat n.mi.

The above noise estimates are based on a nacelle configuration which features an acoustically treated inlet (sonic for approach) and the DAC integrated exhaust system including a jet noise suppressor consisting of a 24-lobe mixer and an acoustically treated ejector. The suppressor area ratio of 3.0 and precise engine operating conditions, aircraft velocity, and altitude are the basis for these estimates. The noise levels are based on exit velocities for a well ventilated ejector nozzle utilizing empirical loss coefficient correlations.

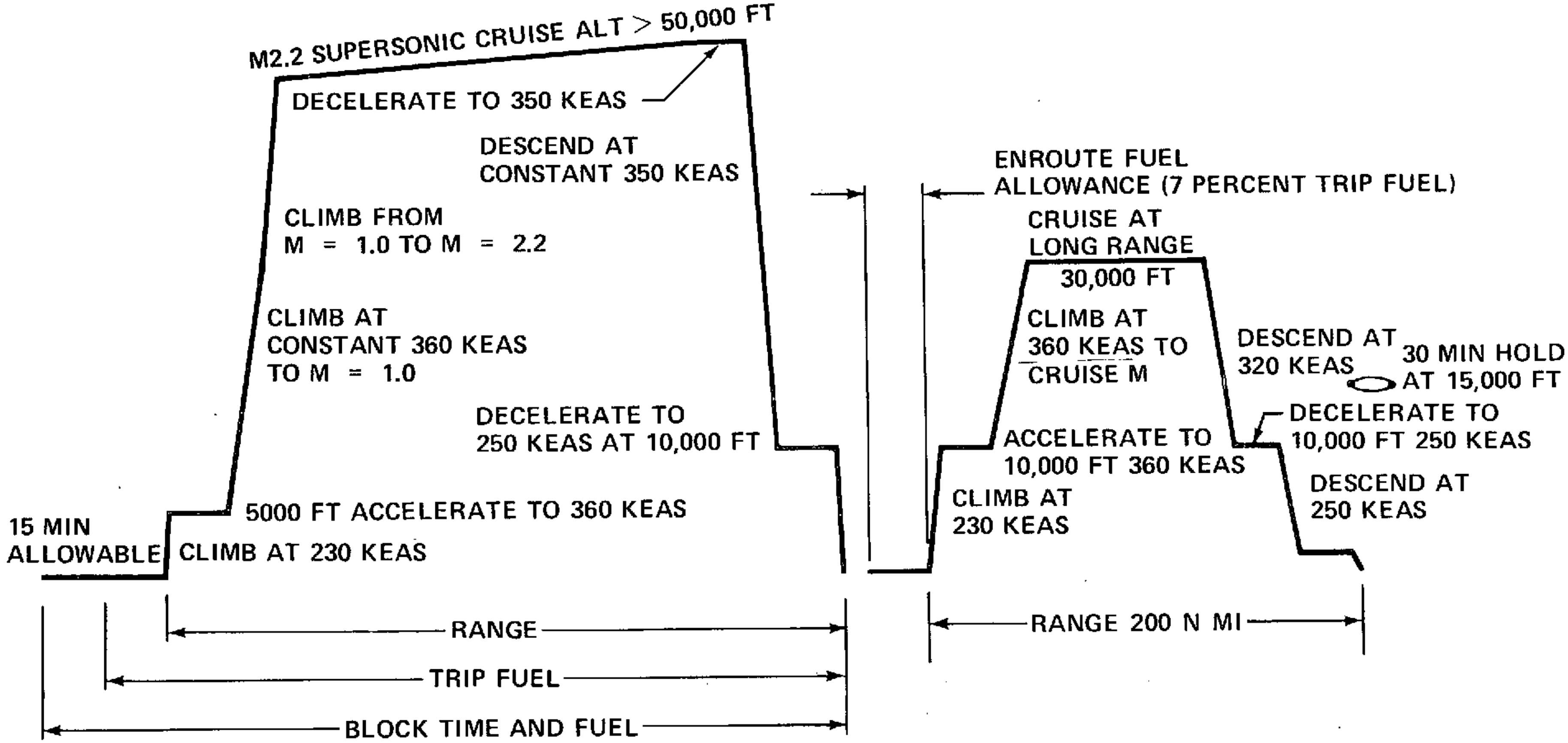


FIGURE 1-20. MISSION PROFILE

The variation in range versus initial subsonic leg length is shown in Figure 1-21. For a 600 n.mile (1110 km) initial subsonic leg, the range penalty is 7 percent.

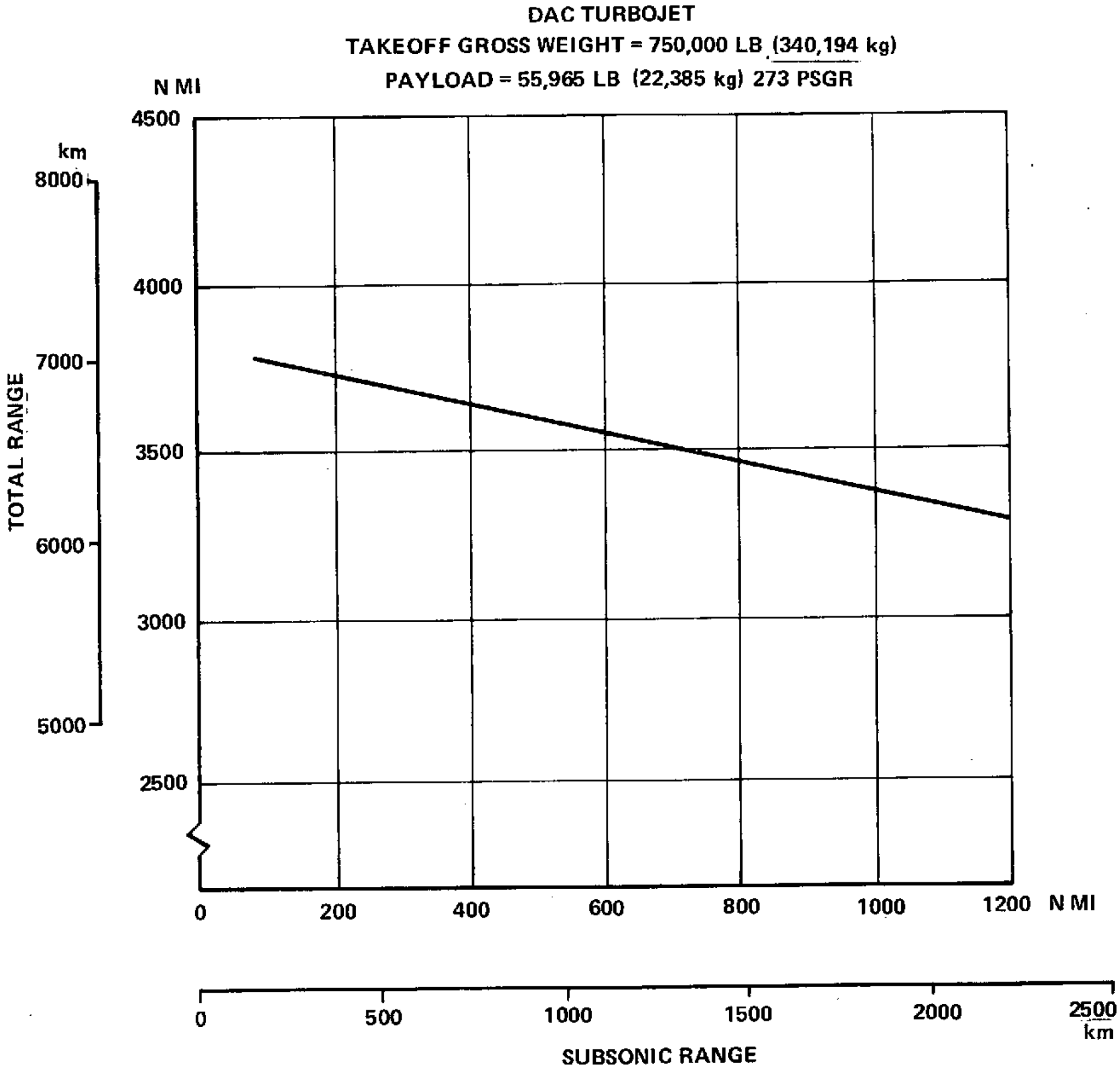


FIGURE 1.21. EFFECT OF INITIAL SUBSONIC LEG ON RANGE

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ENGINE CYCLE SELECTION

General Analysis

During the 1973 NASA AST technology studies, the nonaugmented mini-bypass turbojet, as offered by General Electric was shown to be one of the preferred engine cycles for a Mach 2.2 cruise aircraft. An evaluation has been conducted, utilizing GE data furnished per DAC request to determine relative sensitivity to major cycle parameters and to identify the preferred mini-bypass engine for analysis and airplane integration.

A direct comparison is made between the mini-bypass engine and the DAC non-augmented zero bypass turbojet. The DAC engine had been developed to support the aero sizing and mission performance analyses conducted in the 1973 NASA technology studies. Design turbine inlet temperature of the DAC engine is 2600°F (1700°K) and its compressor pressure ratio is 18:1. A description of this engine is presented in Section 1.

The mini-bypass engine data utilized for this evaluation is based primarily on a matrix of mini-bypass turbojet engines, furnished by GE per DAC request. This engine matrix identified as GE21/J3 Study A2 (P1-P7) includes the following cycle parameter combinations:

Engine No.	CPR	T_{41} (1) °F (°K)
P1	18	2400 (1590)
P2	22	2400 (1590)
P3	25	2400 (1590)
P4	25	2800 (1810)
P5	18	2600 (1700)
P6	22	2600 (1700)
P7	25	2600 (1700)

The approach used for this engine evaluation was to adapt each engine to a baseline airframe/engine configuration and determine the relative performance

(1) T_{41} is defined by GE as design rotor inlet temperature, nominally 200°F (111°K) lower than turbine inlet temperature.

for a defined mission. Each study engine is sized and then installed in a 1973 study airplane configuration, with propulsion system weights and drags adjusted to account for variations in engine size, weight and geometry from the baseline engine.

The evaluation procedure utilizes trade factors that account for differences in climb thrust, climb SFC, supersonic cruise SFC, subsonic SFC's (as defined by reserve requirements), engine size and weight, and engine cost. A computerized technique is used that adapts each of the study engines to the baseline airplane, analyzes their relative performance along the defined mission and calculates overall relative DOC.

The thrust sizing constraint for all the engines is 48,000 lb. takeoff thrust, suppressed, uninstalled at sea level, 0.3 Mach, 86°F (30°C) day, with exhaust jet sideline noise suppressed to FAR Part 36 levels. The GE supplied takeoff performance data for the mini-bypass turbojets are suppressed to FAR Part 36 noise levels and include GE established suppressor losses (nominally 5 percent in net thrust) and cutback requirements commensurate with a 1973 GE defined 10 PNdB jet noise suppressor. For reference, the takeoff performance for the DAC turbojet reflects suppression to FAR Part 36 noise levels, including throttle cut to a maximum exhaust gas temperature of 1500°F (1089°K) (selected as a suppressor material environment limit) and a suppressor loss of approximately 8 percent in net thrust, commensurate with a DAC defined 12 PNdB jet noise suppressor.

A constraint that is considered in the evaluation procedure is maximum climb thrust at end-of-climb/start-of-cruise. Optimum cruise altitude for start-of-cruise is determined for minimum cruise fuel flow, based on thrust/SFC characteristics of the individual engines. Climb performance is then evaluated at start-of-cruise altitude, unless optimum cruise altitude occurs above 52,000 feet (15,850 m). Climb performance is then evaluated at 52,000 feet (15,850 m).

Minimum relative DOC is determined for each engine. In all cases, minimum relative DOC occurs at the minimum sized engines which, for the study, are the engines sized for the FAR Part 36 sideline suppressed takeoff thrust. Features of the sized study engines are presented in Table 2-1.

Figure 2-1 illustrates relative DOC as a function of the primary cycle parameters (design rotor inlet temperature and overall compressor pressure ratio). The minimum DOC solution is shown to favor an engine cycle with high overall compressor pressure ratio (25) and a design rotor inlet temperature of 2600°F (1700°K). Higher DOC is shown for engine cycles having the combination of low design rotor inlet temperature of 2400°F (1589°K) and high overall compressor pressure ratios (22, 25). These cycles are deficient in climb thrust and as a result are penalized by being forced to cruise at altitudes significantly below the optimum cruise altitude. To further illustrate this deficiency, Figure 2-2 presents specific thrust versus cycle parameters at Mach 2.12 climb thrust. Superimposed on this figure is a shaded area encompassing the engine cycles, or cycle parameter combinations, that are identified in Figure 2-1 as climb thrust deficient. The identified cycles are those having low rotor inlet temperature and high compressor pressure ratio and they are shown to have low specific thrust at climb.

TABLE 2-1
BYPASS TURBOJETS

GE DATA

TABLE OF FEATURES

SIZED FOR FAR 36 AT TAKEOFF PER D-3230 -2.2 -4 REQUIREMENTS								
GE2I/J3 STUDY A2	P_1	P_2	P_3	P_4	P_5	P_6	P_7	DAC TJ
ENGINE FEATURES CYCLE – OPR/T41*-/°F (°K)	18/2400 (1587)	22/2400 (1589)	25/2400 (1589)	25/2800 (1811)	18/2600 (1700)	22/2600 (1700)	25/2600 (1700)	18/2600 (1700)
AIRFLOW – LB/SEC (kg/SEC)	773 (350.6)	772 (350.2)	792 (359.3)	771 (349.7)	772 (350.2)	772 (350.2)	773 (350.6)	772 (350.2)
LENGTH, OVERALL – IN. (m)	354 (8.992)	358 (9.093)	371 (9.423)	336 (8.534)	346 (8.788)	348 (8.839)	345 (8.763)	341 (8.561)
DIAMETER, MAX – IN. (m)	78 (1.981)	77 (1.956)	78 (1.981)	74 (1.880)	76 (1.930)	76 (1.930)	75 (1.905)	82 (2.083)
WEIGHT – LB** (kg)	13,470 (6110)	13,990 (6345.9)	14,850 (6736)	13,170 (5973.9)	13,020 (5905.9)	13,470 (6110)	13,720 (6223.4)	15,640 (7094.3)
ESTIMATED COST (1972 $M)	2.62	2.66	2.70	2.88	2.67	2.74	2.79	2.69

*T41 = DESIGN ROTOR INLET TEMPERATURE
**INCLUDES NOZZLE AND SUPPRESSOR
***2600°F (1700°K) TURBINE INLET TEMPERATURE FOR DAC TJ

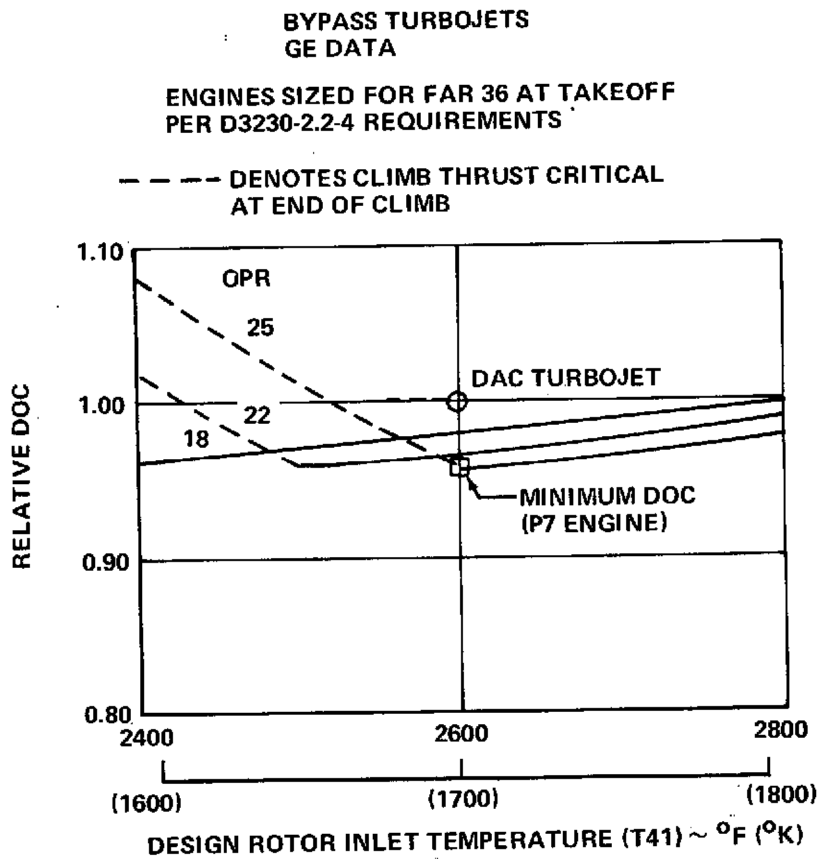


FIGURE 2-1. CYCLE VARIABLE EVALUATION

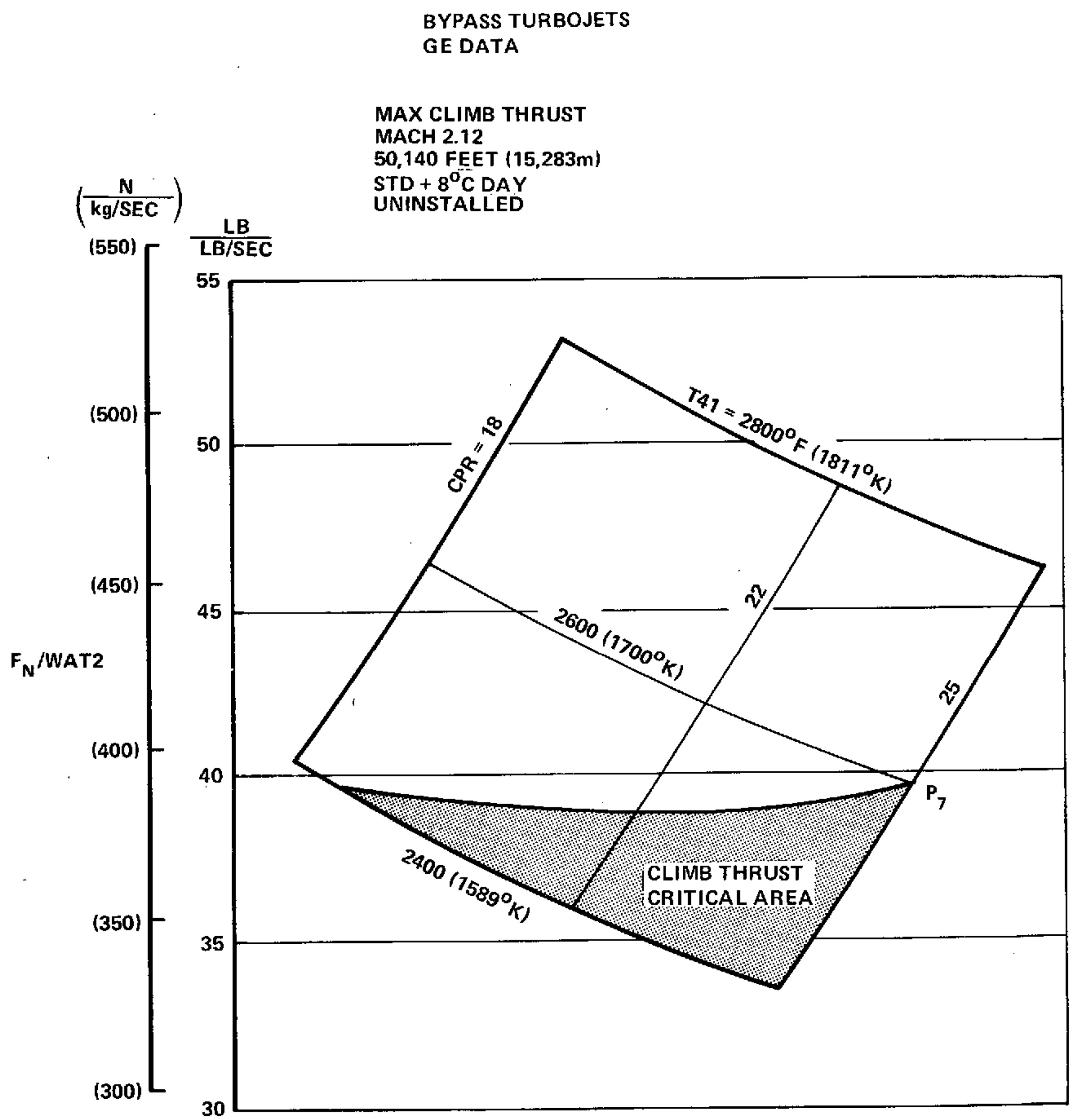


FIGURE 2-2. ESTIMATED SPECIFIC THRUST AT CLIMB

The climb thrust critical engines, however, exhibit superior SFC characteristics. Figures 2-3, 2-4 and 2-5 show SFC versus cycle parameters at Mach 2.12 supersonic cruise, 50,140 feet (15,283 m), at Mach 0.95 subsonic cruise 35,000 feet (10,668 m) and at Mach 0.4, 1500 feet (457 m), respectively. Superimposed on each of these figures is a shaded area representing the climb thrust critical cycle parameter combinations identified in Figure 2-2. It is shown that the cycles in this climb thrust critical area are those offering the best SFC's at all three cruise flight conditions.

The performance characteristics of all the engines in the study engine matrix are summarized in Table 2-2. It is noted that engine P7, identified in Figure 2-1 as the cycle resulting in the lowest relative DOC solution, is the engine with the lowest cruise SFC's that is not climb thrust critical.

For reference, the minimum relative DOC solution for the baseline configuration is shown in Figure 2-1. The P7 shows a 4 percent reduction in relative DOC from the DAC turbojet. This difference is attributable mainly to the greater weight of the DAC turbojet [15,640 lb. (7,094 kg), as compared to 13,720 lb. (6,223 kg)], and, to a lesser degree, higher SFC's at subsonic cruise and loiter. The DAC baseline turbojet performance characteristics are summarized in Table 2-2 for comparison with the mini-bypass engine matrix.

Conclusion

The P7 cycle was identified as the optimum mini-bypass turbojet for further study. It is the cycle used for the in-depth mini-bypass turbojet engine/ D3230-2.2-5B airplane integration studies summarized in this report. NASA concurred in this selection.

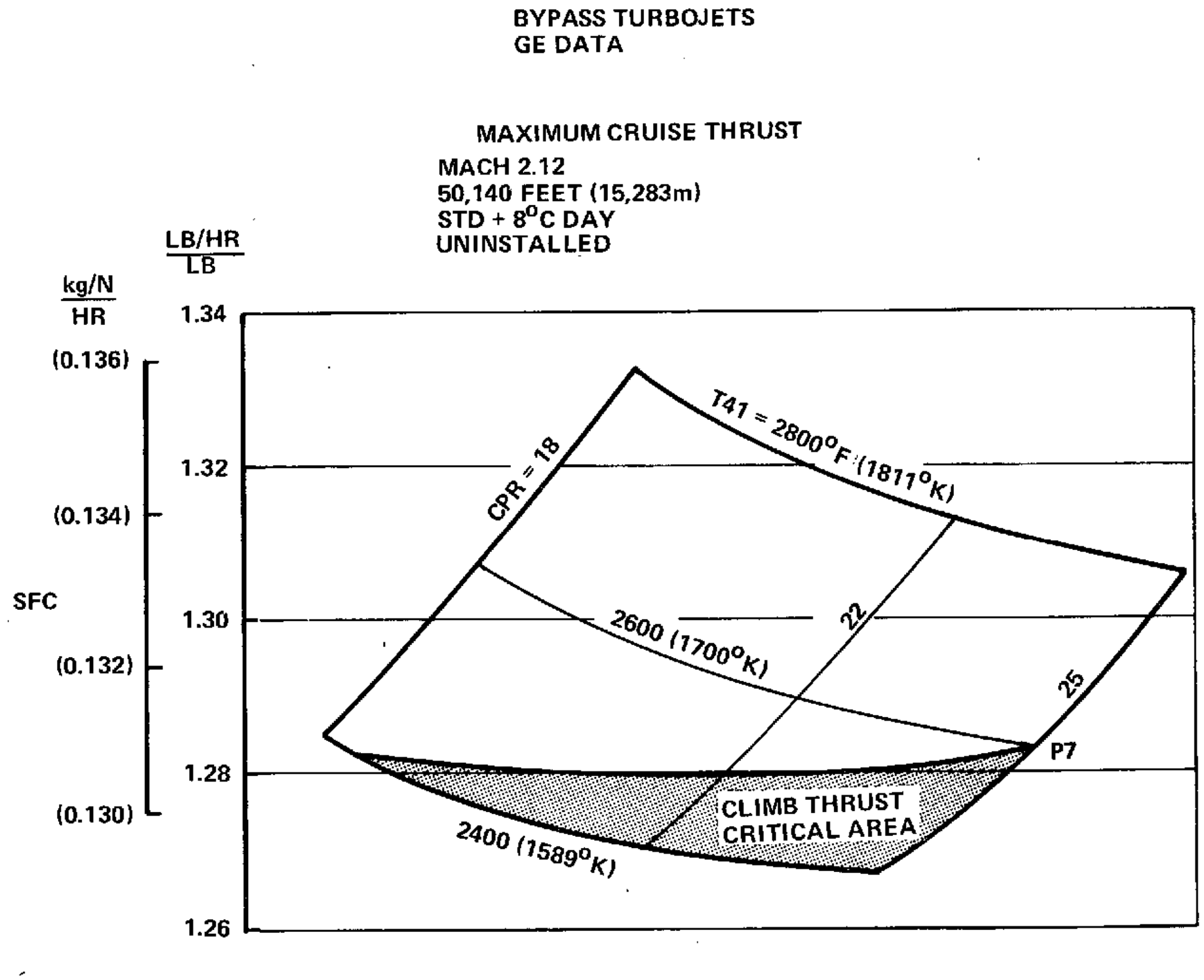


FIGURE 2-3. ESTIMATED SFC AT SUPERSONIC CRUISE

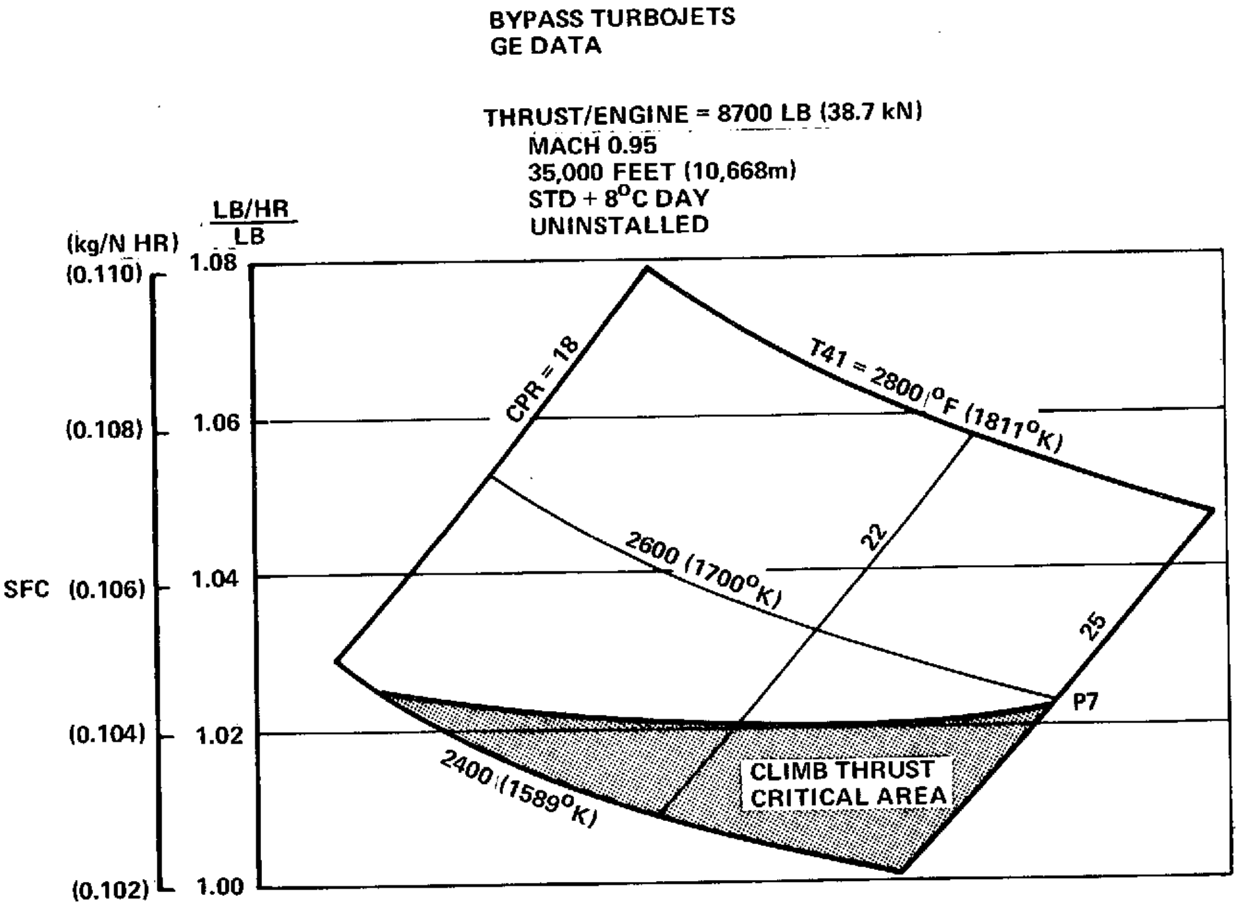


FIGURE 2-4. ESTIMATED SFC AT SUBSONIC CRUISE

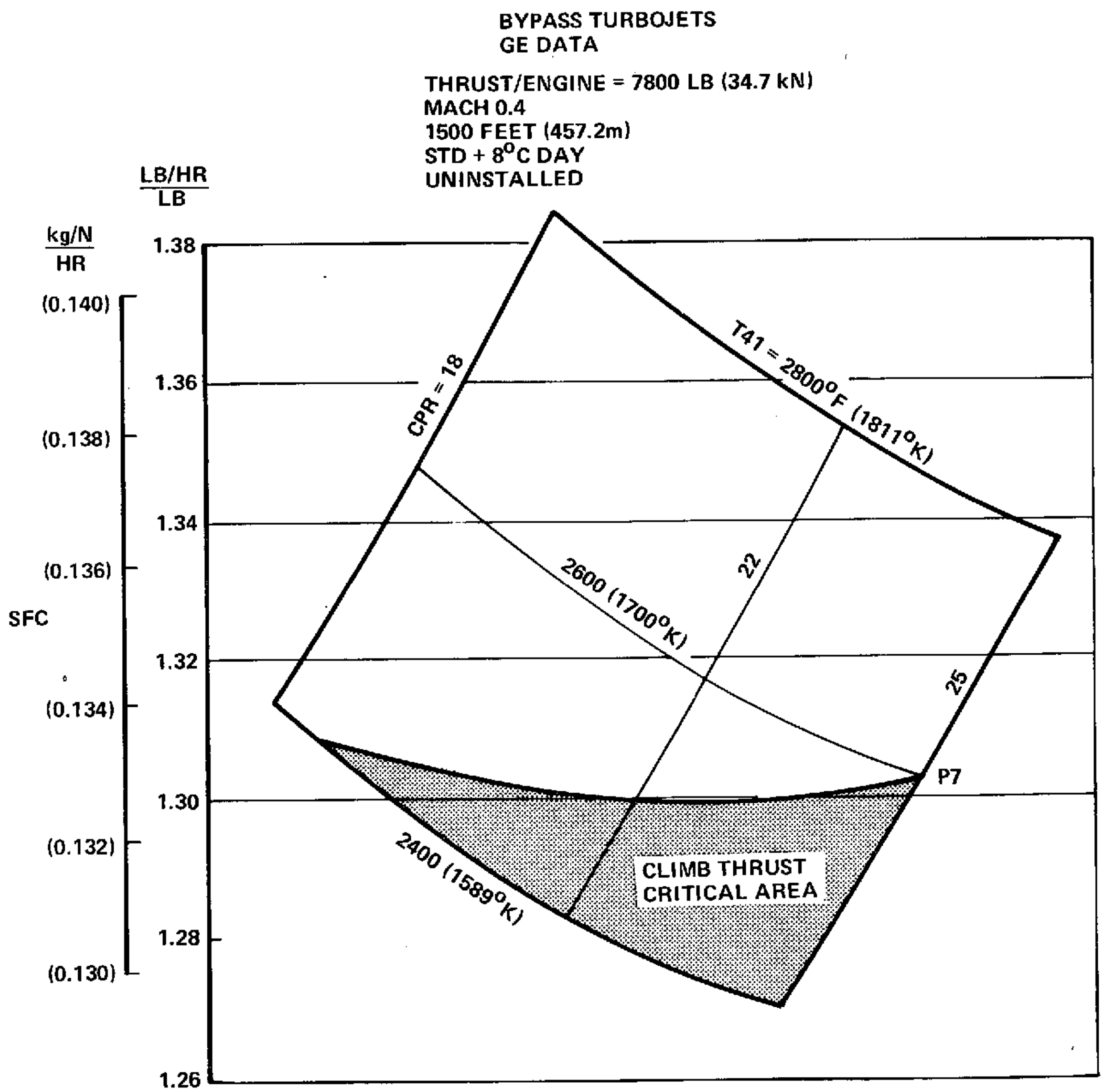


FIGURE 2-5. ESTIMATED SFC AT LOITER

TABLE 2-2

BYPASS TURBOJETS G.E. DATA TABLE OF PERFORMANCE

SIZED FOR FAR 36 AT TAKEOFF PER D-3230 -2.2-4 REQUIREMENTS

GE21/J3 STUDY A2	P_1	P_2****	P_3****	P_4	P_5	P_6	P_7	DAC TJ
CYCLE – OPR/T41*-/°F (°K)	18/2400 (1589)	22/2400 (1589)	25/2400 (1589)	25/2800 (1811)	18/2600 (1700)	22/2600 (1700)	25/2600 (1700)	18/2600*** (1700)
ENGINE PERFORMANCE**								
TAKEOFF THRUST – LB (kN) [SUPP, S.L., 3M, 86°F (30°C) DAY]	48,000 (213.5)	48,000 (213.5)	48,000 (213.5)	48,000 (213.5)	48,000 (213.5)	48,000 (213.5)	48,000 (213.5)	48,000 (213.5)
MAX CRUISE THRUST – LB [2.12M, 50,140 FT (15,283m)]	21,080 (93.77)	18,700 (83.18)	17,860 (79.44)	22,580 (100.44)	23,000 (102.31)	21,610 (96.13)	20,190 (89.81)	24,165 (107.49)
MAX CRUISE SFC – LB/HR/LB (kg/HR/N) [2.12M, 50,140 FT (15,283m)]	1.29 (0.132)	1.27 (0.130)	1.27 (0.130)	1.31 (0.134)	1.31 (0.134)	1.29 (0.132)	1.28 (0.131)	1.28 (0.131)
SUBSONIC CRUISE SFC – LB/HR/LB (kg/HR/N) [0.95M, 35,000 FT (10,668m) 8700 LB (38.7 kN) THRUST]	1.03 (0.105)	1.01 (0.103)	1.0 (0.102)	1.06 (0.108)	1.05 (0.107)	1.03 (0.105)	1.02 (0.104)	1.05 (0.107)
LOITER CRUISE SFC – LB/HR/LB (kg/HR/N) [0.4M, 1500 FT (457.2m) 7800 LB (34.7 kN) THRUST]	1.31 (0.134)	1.28 (0.131)	1.27 (0.130)	1.34 (0.137)	1.35 (0.138)	1.32 (0.135)	1.30 (0.133)	1.68 (0.171)

*T41 – DESIGN ROTOR INLET TEMPERATURE
**UNINSTALLED
***2600°F (1700°K) TURBINE INLET TEMPERATURE FOR DAC TJ
****CLIMB THRUST CRITICAL

ENGINE SIZING

General Analysis

Engine data for sizing and performance are based on the GE P7 engine, identified in the previous section as the preferred mini-bypass turbojet cycle.

An engine size is identified based on sizing criteria commensurate with the -5A AST configuration defined in Section 1. The sizing criteria considered for this engine are takeoff (both sideline and takeoff/cutback) for FAR Part 36 noise levels and start-of-cruise altitude. Figure 2-6 illustrates the engine sizing logic.

First, start-of-cruise altitude is shown as a function of engine size. For initial sizing, start-of-cruise altitude was established at 56,000 ft. (17,069 m). However, the engine size required for start-of-cruise at this altitude appeared excessive, over 900 lb/sec (408 kg/sec). Subsequently, mission trade studies were conducted to observe the effect of varying engine size on mission performance and to determine if selection of a smaller engine size could be justified. Results of these trade studies indicate that maximum mission range occurs at an engine size of approximately 782 lb/sec (355 kg/sec) with a start-of-cruise altitude of 52,000 ft. (15,850 m). Therefore, using mission range as the figure of merit, an engine size of 782 lb/sec (355 kg/sec) has been selected.

This engine size is now examined against takeoff requirements, considering FAR Part 36 noise levels, in-house estimated jet noise characteristics and GE defined suppressor technology. The suppressor design point envelope shown in Figure 2-6 is a locus of the capability of design suppressors supplied by GE consistent with the level of technology identified with this study activity. The suppressor off-design characteristics shown is identified by GE as consistent with the technology level of this study activity.

GE MINI-BYPASS
GE SUPPRESSOR DATA USED FOR SIZING
D-3230-2.2-5

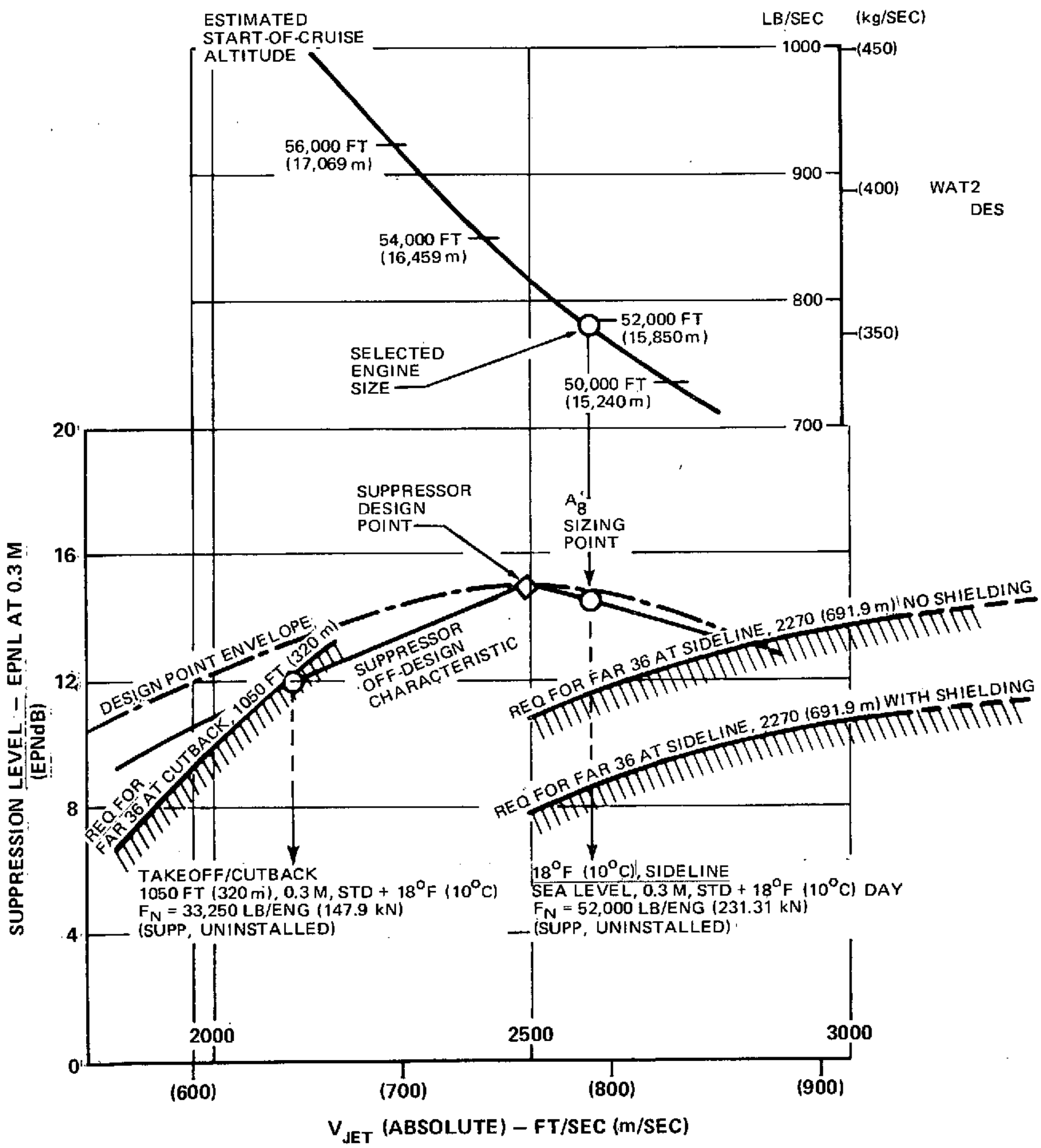


FIGURE 2-6. ENGINE SIZING FOR TAKEOFF/CRUISE

The selected suppressor has a design point jet velocity of 2500 ft/sec (762 m/sec), operating off-design at 2600 ft/sec (793 m/sec) for sideline. A schematic of the exhaust system including the selected suppressor is shown in Figure 2-7. With the suppressor deployed, the fixed nozzle throat area is established at the sideline condition. (A8 sizing point on Figure 2-6.) The suppressed uninstalled thrust available at this condition is 52,000 lb/engine (231.3 kN), equal to the takeoff thrust requirements identified for the -5A baseline airplane. The gross thrust loss identified with this suppressor at takeoff is approximately 4.4 percent.

DAC estimated engine exhaust jet noise suppression requirements for FAR Part 36 are shown in Figure 2-6 for sideline [2270 ft. (692 m) sideline distance, sea level, 0.3 Mach, Std. + 18°F (10°C) day]. Suppressor performance is shown to be more than adequate, exceeding suppression requirements for FAR Part 36 by an estimated 3 EPNdB.

Suppression requirements versus suppressor performance are further examined at takeoff/cutback [1050 ft. (320 m) altitude, 0.3 Mach, Std. + 18°F (10°C) day]. At this condition, the throttle is cut back to a thrust level of 33,250 lb/engine (147.9 kN), identified as the thrust required to meet the 4 percent climb gradient for the -5A baseline airplane. At this condition, suppressor performance is predicted to meet DAC estimated suppression requirements for FAR Part 36.

The engine size of 782 lb/sec (355 kg/sec) has been selected as it combines maximum range potential with noise levels commensurate with FAR Part 36 (with GE identified suppressor technology of 15 PNdB). Therefore, it is the engine size utilized for configuration integration and initial mission studies.

GE MINI-BYPASS

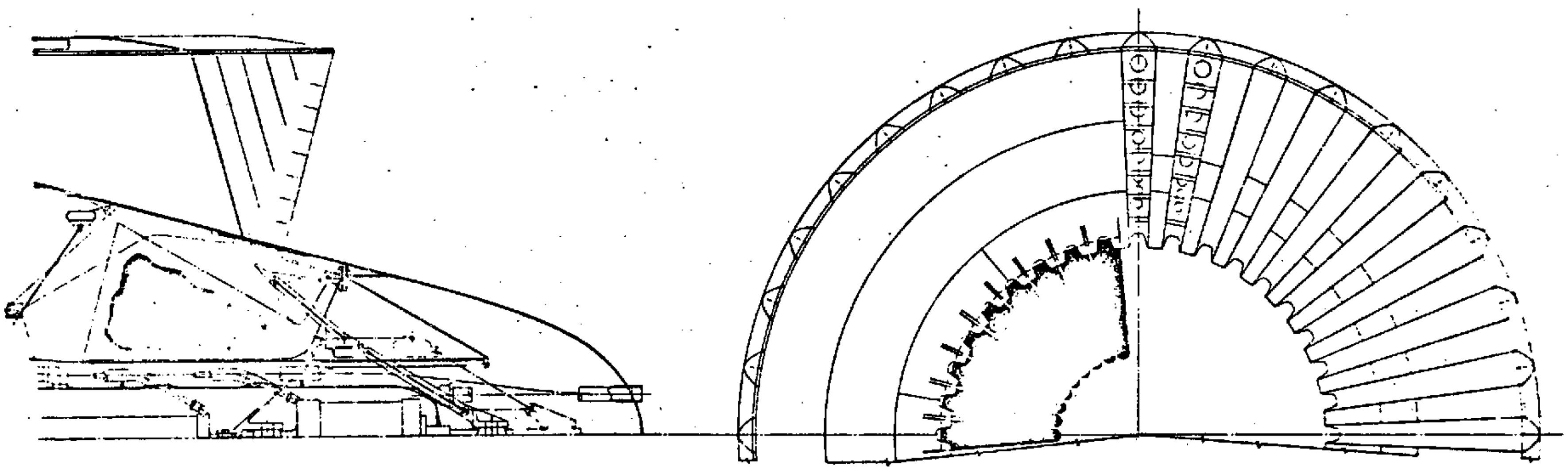


FIGURE 2-7. EXHAUST SYSTEM SUPPRESSOR SCHEMATIC

Engine Definition

The engine is a twin spool, non-augmented, mini-bypass turbojet which is designed for Mach 2.2 supersonic cruise operation and incorporates a technology level commensurate with a 1978 go-ahead (Figure 2-8). It is sized for an inlet corrected airflow of 782 lb/sec (355 kg/sec) at takeoff rating for sea level static, standard day. The design cycle characteristics and ratings are shown in Table 2-3. The exhaust system is an annular, translating shroud, convergent divergent plug configuration. The nozzle has a variable geometry throat for thrust modulation and a translating cylindrical shroud to provide the internal area for expansion of the exhaust gases. Cooling of the nozzle is provided by LP compressor discharge air. No secondary airflow is required for cooling purposes and no provisions are incorporated to handle secondary airflow from the intake duct. Thrust reversing is achieved by diverting the exhaust gas flow through a series of cascades mounted in the nozzle shroud. The jet noise suppressor acting on the exhaust stream is a 36-chute type configuration which is positioned aft of the variable nozzle throat mechanism. During the suppressed mode, the variable nozzle throat is collapsed to the furthermost open position and the nozzle throat is then formed by the deployed suppressor elements. During unsuppressed operation the suppressor is fully stowed within the nozzle plug. A sketch of the engine is shown in Figure 2-9. The installed engine is shown in Figure 2-10.

Engine weights, dimensions, scaling equations and cost data are presented in Table 2-3. Cost data are based on GE cost information provided as part of their Advanced Supersonic Propulsion System Technology studies conducted under contract to NASA Lewis. Costs have been escalated to 1973 by DAC based on 1972 dollar values provided by the engine manufacturers' study.

GE MINI-BYPASS

GE 21/P7

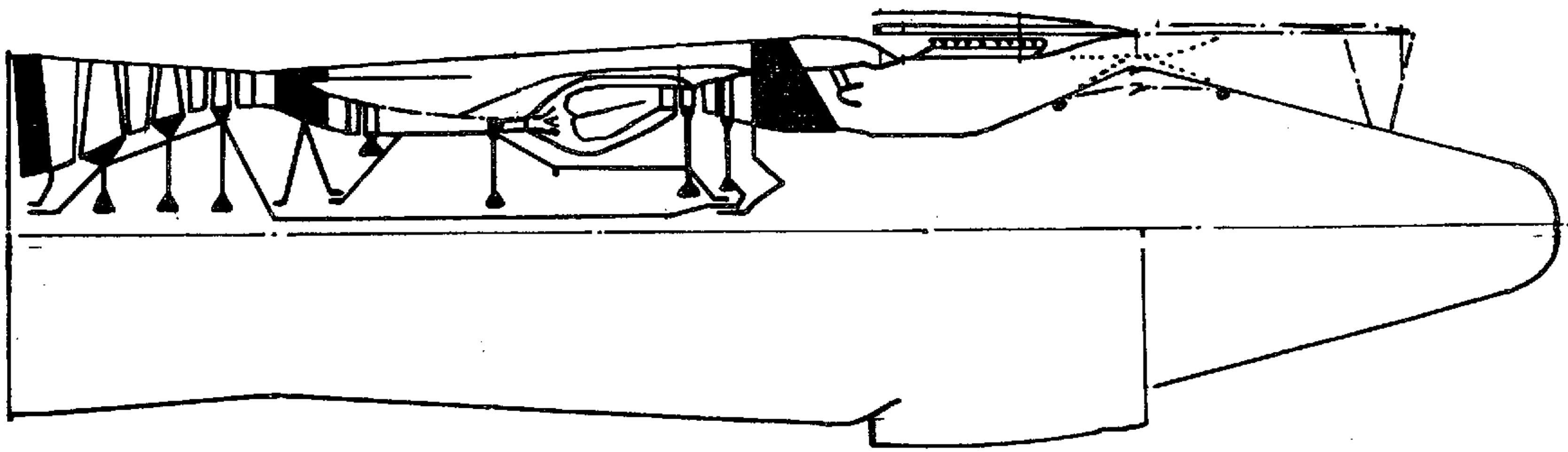


FIGURE 2-8. ENGINE SCHEMATIC

TABLE 2-3
GE MINI-BYPASS ENGINE CHARACTERISTICS SUMMARY
782 LB/SEC (355 kg/SEC) RATED AIRFLOW

DESIGN CYCLE CHARACTERISTICS

BYPASS RATIO	0.1
LP SPOOL PRESSURE RATIO	3.35
CYCLE PRESSURE RATIO	24.7
T_{41}* (TAKEOFF, MAX CLIMB, MAX CRUISE)	2600°F (1700°K)

TAKEOFF RATINGS [STD DAY + 18°F (10°C)]

MAX THRUST (SLS) –	LB (kN)	74,700 (332.28)
MAX THRUST (SL, 0.3M, UNINSTALLED) –	LB (kN)	70,000 (311.37)
THRUST AT 2600 FPS (792.5 M/S) (SL, 0.3M, UNINSTALLED) (1270°F EGT) –	LB (kN)	54,300 (241.54)

WEIGHT

BASE ENGINE	– LB (kg)	11,469	(5202.3)
NOZZLE	– LB (kg)	1,764	(800.2)
REVERSER	– LB (kg)	460	(208.7)
SUPPRESSOR	– LB (kg)	720	(326.6)
TOTAL – LB (kg)		14,413	(6537.8)

DIMENSIONS

ENGINE INLET GAS FLOW PATH DIAMETER – IN. (m)	64.5 (1.638)
HUB-TO-TIP RATIO (AT ROTOR INLET)	0.436
HUB-TO-TIP RATIO (AT PLANE OF ATTACH FLANGE)	0.312
ENGINE MAX DIAMETER – IN. (m)	77.7 (1.974)
LENGTH – INLET FLANGE TO EXHAUST PLUG TIP – IN. (m)	283.0 (7.188)

SCALING FACTORS

WEIGHT $W_2 = W_1 \left(\frac{WAT2}{782}\right)^{1.2}$

DIAMETER $D_2 = D_1 \left(\frac{WAT2}{782}\right)^{0.5}$

LENGTH $L_2 = L_1 \left(\frac{WAT2}{782}\right)^{0.5}$

COST**

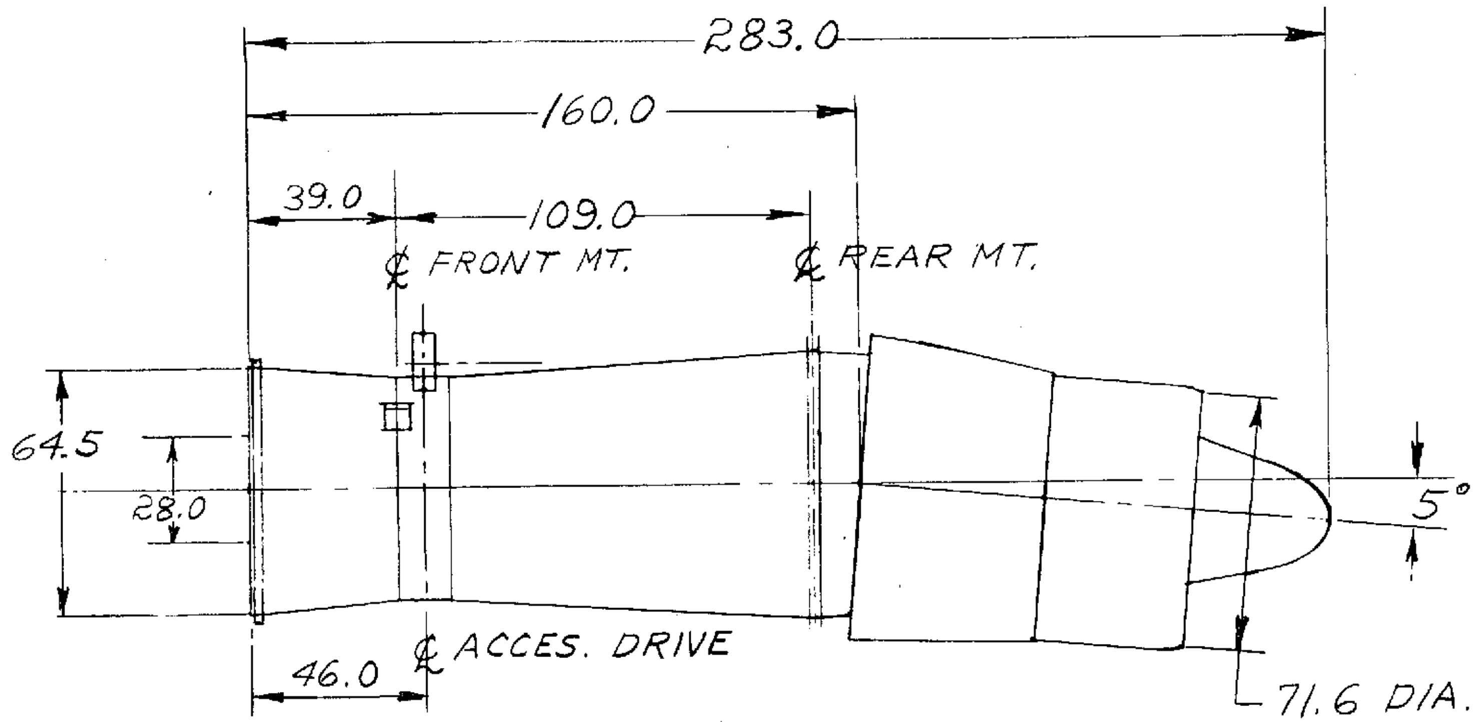
WITH NOZZLE/REVERSER/SUPPRESSOR $3.04M

SCALING FACTOR

$$COST_2 = COST_1 \left(\frac{WAT2}{782}\right)^{0.64}$$

*T_{41} IS DEFINED BY GE AS DESIGN ROTOR INLET TEMPERATURE NOMINALLY 200°F (111°K) LOWER THAN TURBINE INLET TEMPERATURE

**BASED ON
- 1973 DOLLARS
- 1978 ENGINE TECHNOLOGY
- PRICES INCLUDE ALL DEVELOPMENT COSTS PLUS FIVE-YEAR PRODUCT SUPPORT AFTER CERTIFICATION BASED ON ONE ENGINE MODEL
- 3000 ENGINE PRODUCTION RUN



AIRFLOW 782 LBS/SEC (355 kg/SEC)

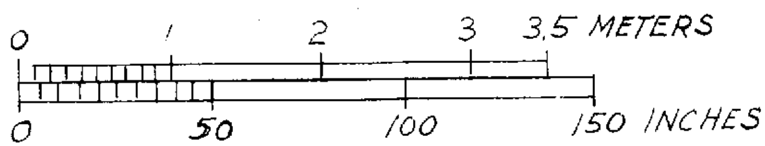


FIGURE 2.9. GE MINI-BYPASS ENGINE

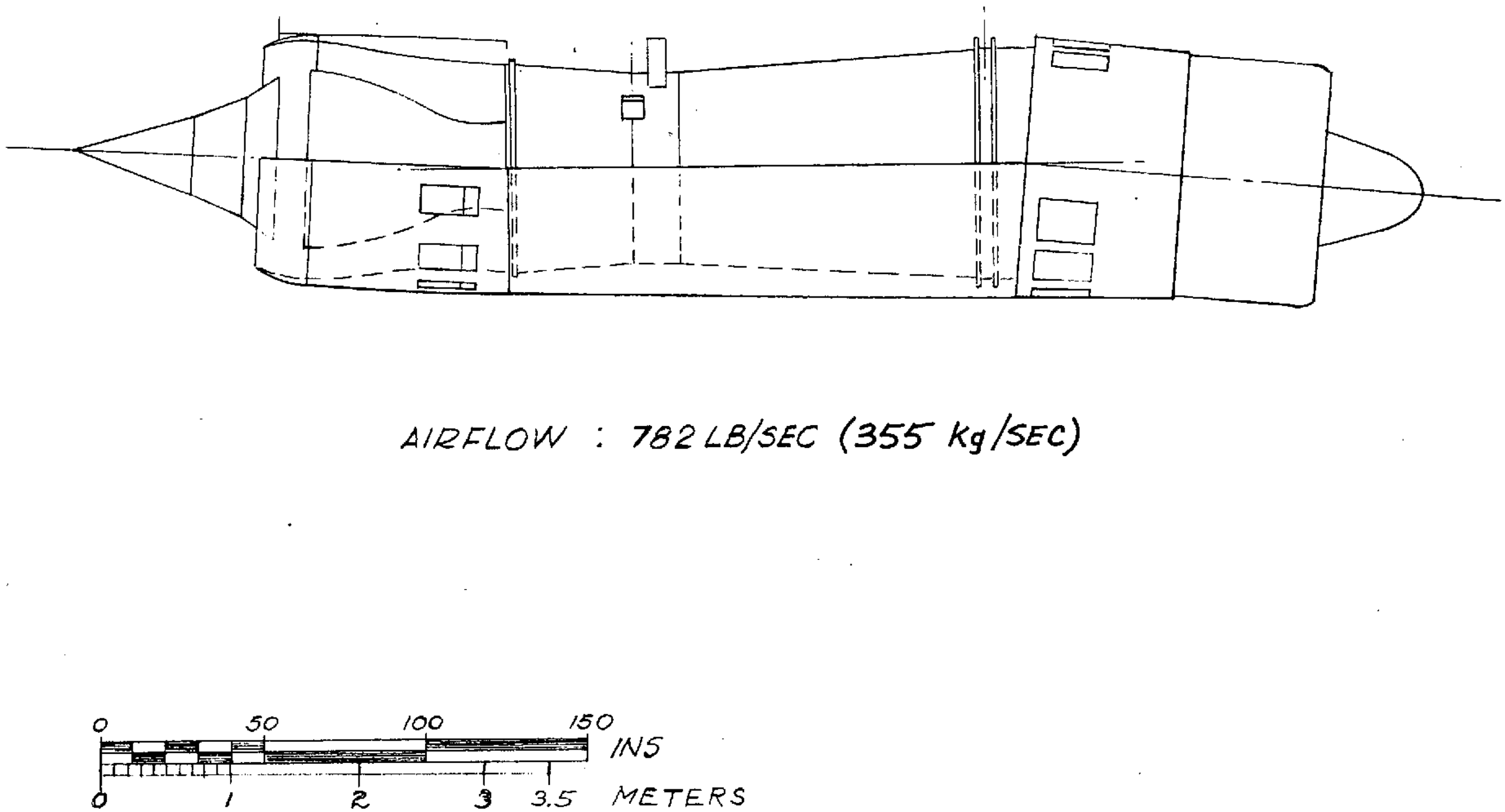


FIGURE 2.10. GE MINI-BYPASS ENGINE INSTALLATION

PROPULSION SYSTEM PERFORMANCE

Uninstalled Performance

At DAC request, GE furnished uninstalled engine performance data at Mach 2.2 with DAC airflow schedule and inlet recovery. The uninstalled engine performance includes the effects of:

° U.S. 1962 model atmosphere	
° Inlet recovery	Figure 1-6
° GE supplied internal nozzle velocity coefficient	
° Customer compressor air bleed	1 lb/sec (.454 kg/sec)
° Customer power extraction	200 HP (149 kW)
° Jet A Fuel, Lower Heating Value	18,400 BTU/lb. (4.34×10^7 J/kg)
° No losses for acoustical treatment	

Installed Performance Analysis

The analysis of the propulsion system performance of the mini-bypass P7 engine follows the same procedures used for the baseline turbojet engine (Section 1). The inlet performance and the nacelle analysis include an evaluation of the following items:

- Inlet spillage drag
- Inlet bypass drag
- Engine and ECS cooling airflow drag
- Nacelle skin friction drag
- Nacelle afterbody drag
- Nacelle wave drag

The inlet geometry and cone schedules are the same as used for the turbojet engine. The inlet total pressure recovery variation is shown in Figure 1-6. Also shown in the figure is the variation of inlet critical mass-flow ratio and the inlet cone schedule. Shown in Figure 1-7 is the mass-flow-ratio for the inlet boundary layer bleed airflow.

The engine airflow schedule for the P7 engine is the same as for the baseline turbojet (see Figure 1-8). The installed inlet performance for the P7 engine is shown in Figure 2-11. As shown by the upper graph in the figure, the inlet airflow supply provides an adequate match with the engine airflow demand. The inlet is sized at the design point of 2.2M. The sized capture area is 24.2 ft^2 (2.25 m^2). The engine and ECS cooling airflow is based on an allowance of 2 percent of inlet capture area airflow for the environmental control system (ECS) cooling and for engine compartment ventilation.

The nacelle drag coefficient buildup is shown in the lower graph in Figure 2-11. The inlet drag characteristics are calculated by combining the mass-flow-ratio characteristics with empirical drag coefficient correlations. For the convenience of engine sizing studies, the nacelle skin friction drag are included in the installed engine performance. The skin friction coefficients are based on fully turbulent flat plate adiabatic wall boundary layer data with transition at the leading edge and the resulting drag is shown.

The nacelle afterbody drag is dependent on the nozzle exit area, pressure ratio, and flight Mach number. The maximum nozzle area is sized at 2.2M, maximum climb thrust. The engine dependent boattail drag at this condition is zero. As nozzle area decreases for lower Mach numbers and reduced power settings, the boattail drag increases. The boattail drag identified with this area change is

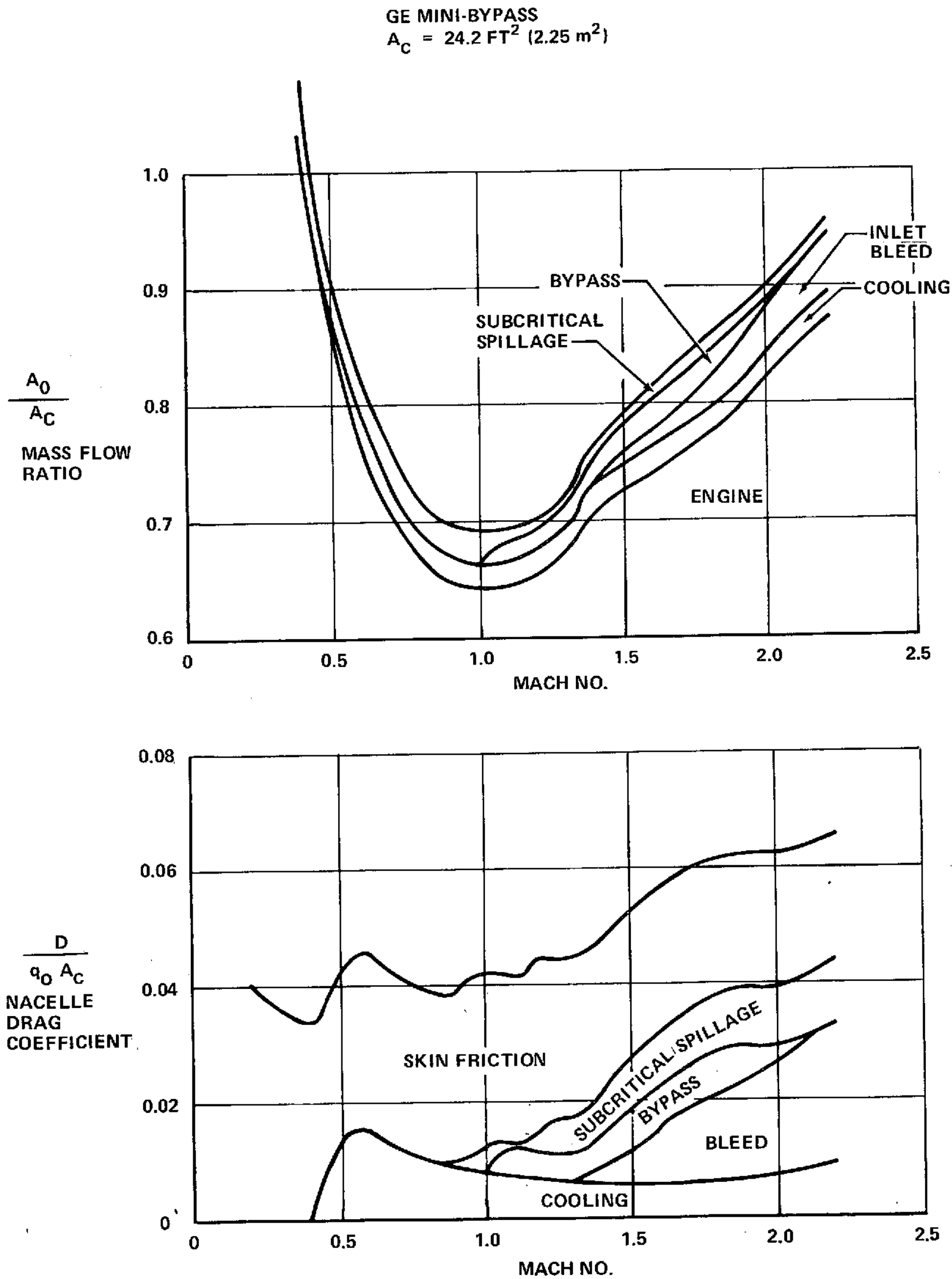


FIGURE 2-11. INSTALLED INLET PERFORMANCE

based on drag characteristics provided by GE for their convergent/divergent plug nozzle configuration (Figure 2-12). The variations in drag coefficient relative to the design condition along the aircraft climb path at maximum climb thrust and for subsonic flight are shown in Figures 2-13 and 2-14.

The nacelle wave drag in the presence of the aircraft, including the supercritical spillage drag and the design afterbody drag is treated as part of the aircraft wave drag.

Performance Results

Installed propulsion system performance is generated by correcting the uninstalled engine performance data for the installation effects described above. The climb performance characteristics are generated along the aircraft flight path shown in Figure 1-11. Uninstalled and installed thrust for the takeoff power setting (EGT limited for noise) are shown in Figure 2-15. Figures 2-16 and 2-17 present the uninstalled and installed referred thrust and SFC, respectively, for maximum climb thrust along the climb flight path. Uninstalled and installed supersonic cruise, subsonic cruise (for alternate mission), and hold performance are shown in Figure 2-18 through 2-20. Figure 2-21 presents the installed characteristics used along the descent flight path.

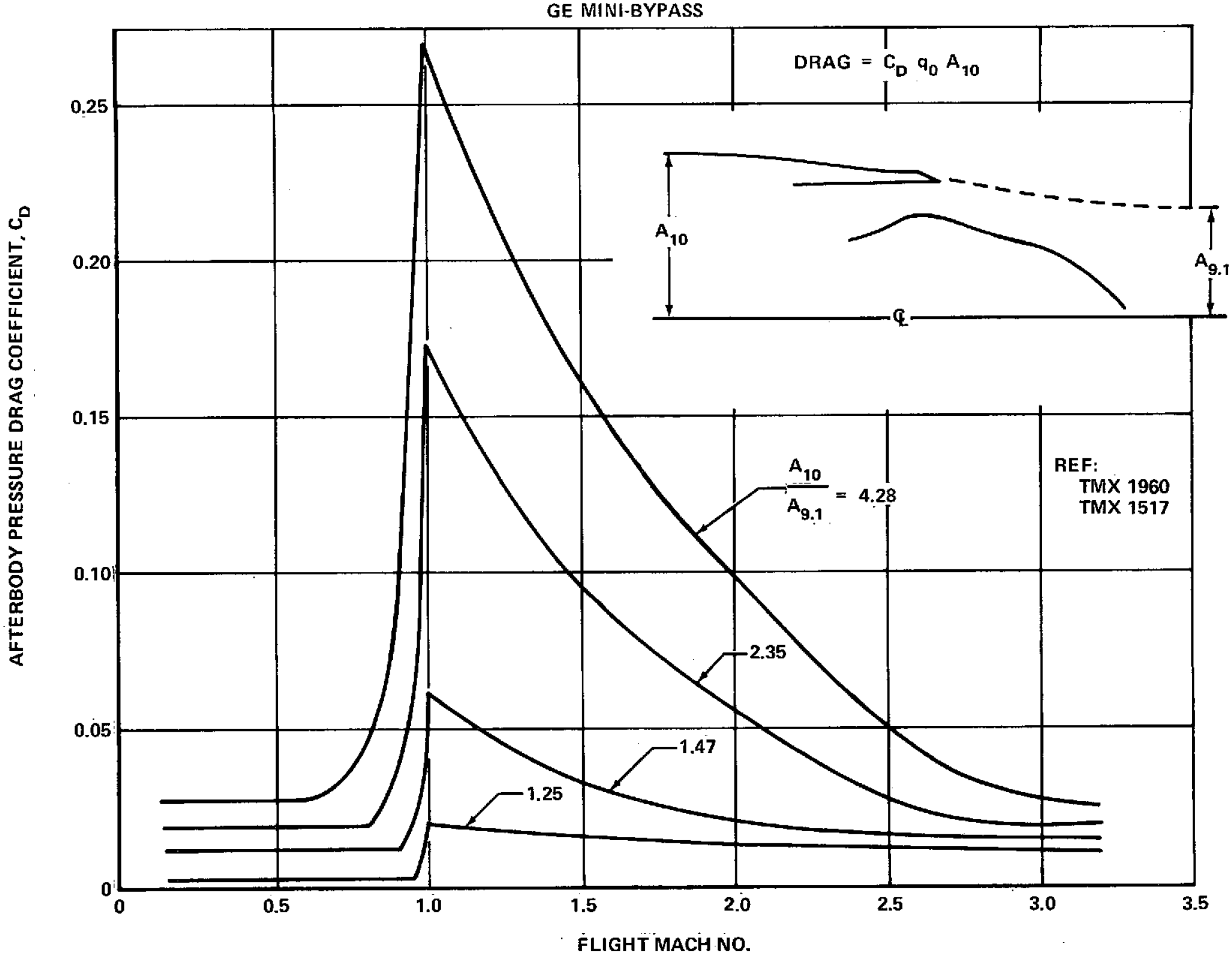


FIGURE 2.12. ESTIMATED AFTERBODY PRESSURE DRAG

GE MINI-BYPASS
STD DAY
A_c = 24.2 FT^2 (2.25 m^2)
WAT2 = 782 LB/SEC (355 kg/SEC)

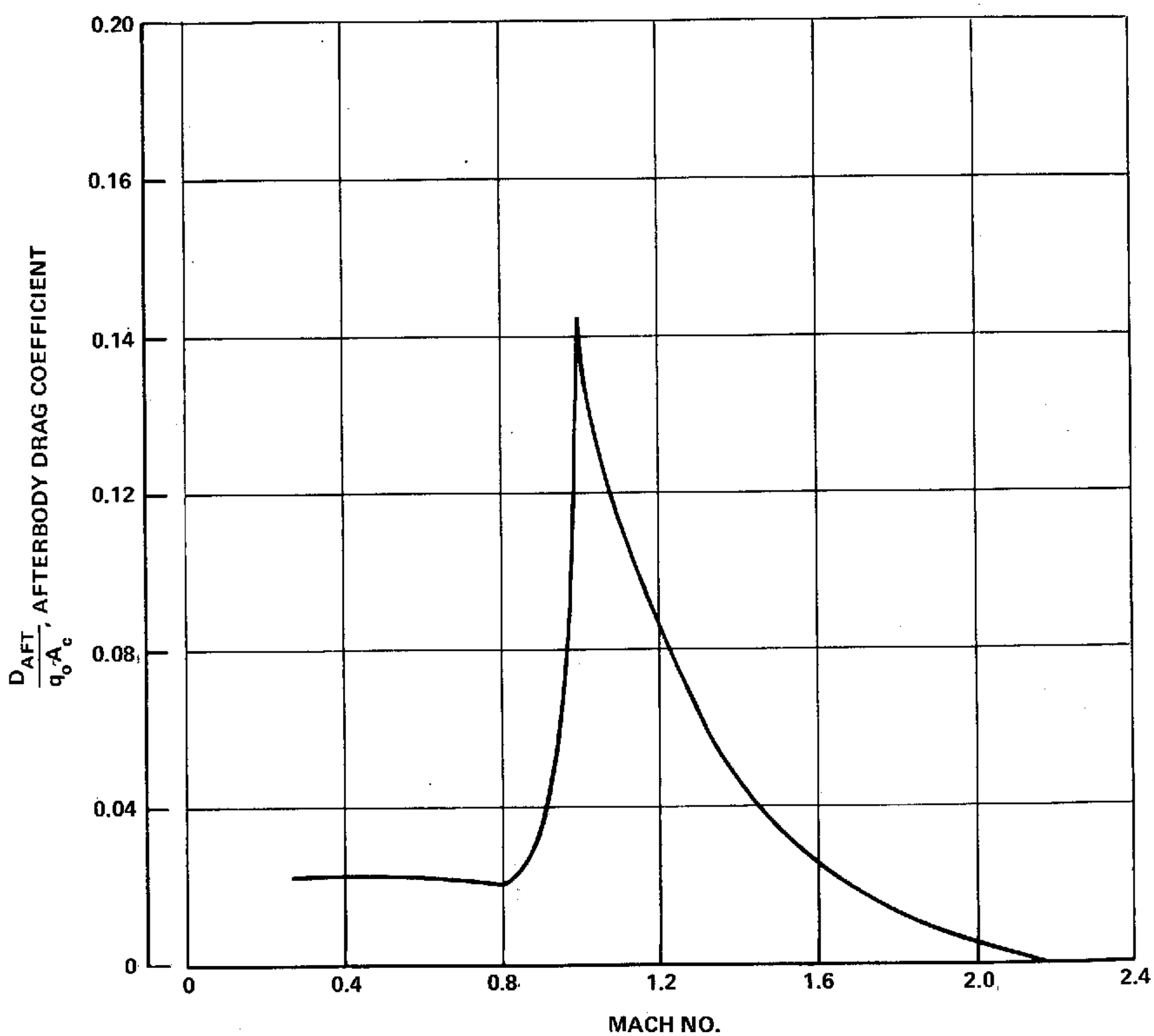


FIGURE 2-13. CLIMB AFTERBODY DRAG

GE MINI-BYPASS
STD DAY
A_C = 24.2 FT^2 (2.25 m^2)
WAT2 = 782 LB/SEC (355 kg/SEC)

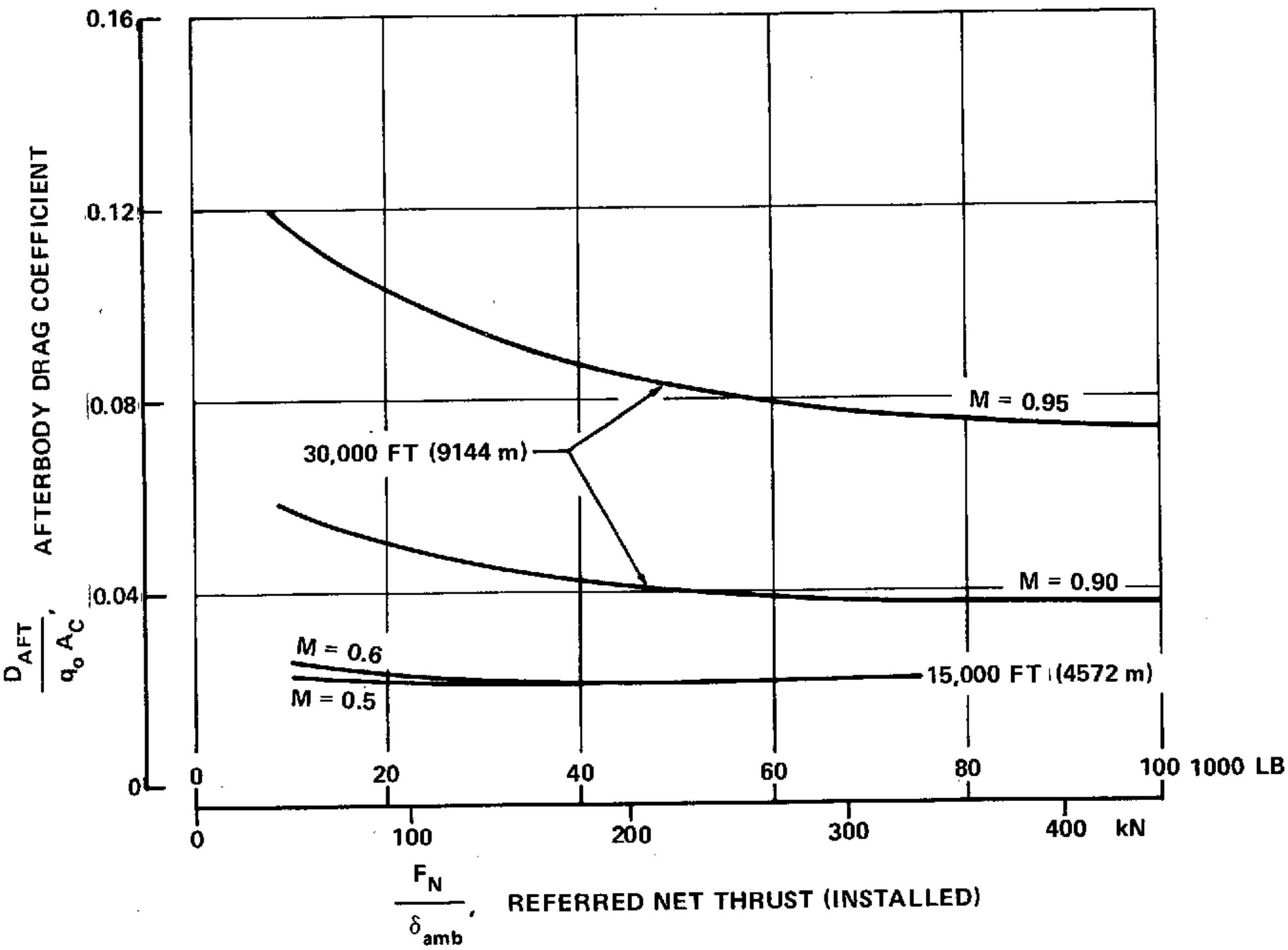


FIGURE 2-14. SUBSONIC AFTERBODY DRAG

GE MINI-BYPASS

SEA LEVEL, STD + 18°F (10°C) DAY
WAT2 = 782 LB/SEC (355 kg/SEC)

SLS RATING = 74,700 LB (332.28 kN)
100% SLS THRUST (UNINSTALLED) = 70,000 LB (311.37 kN)

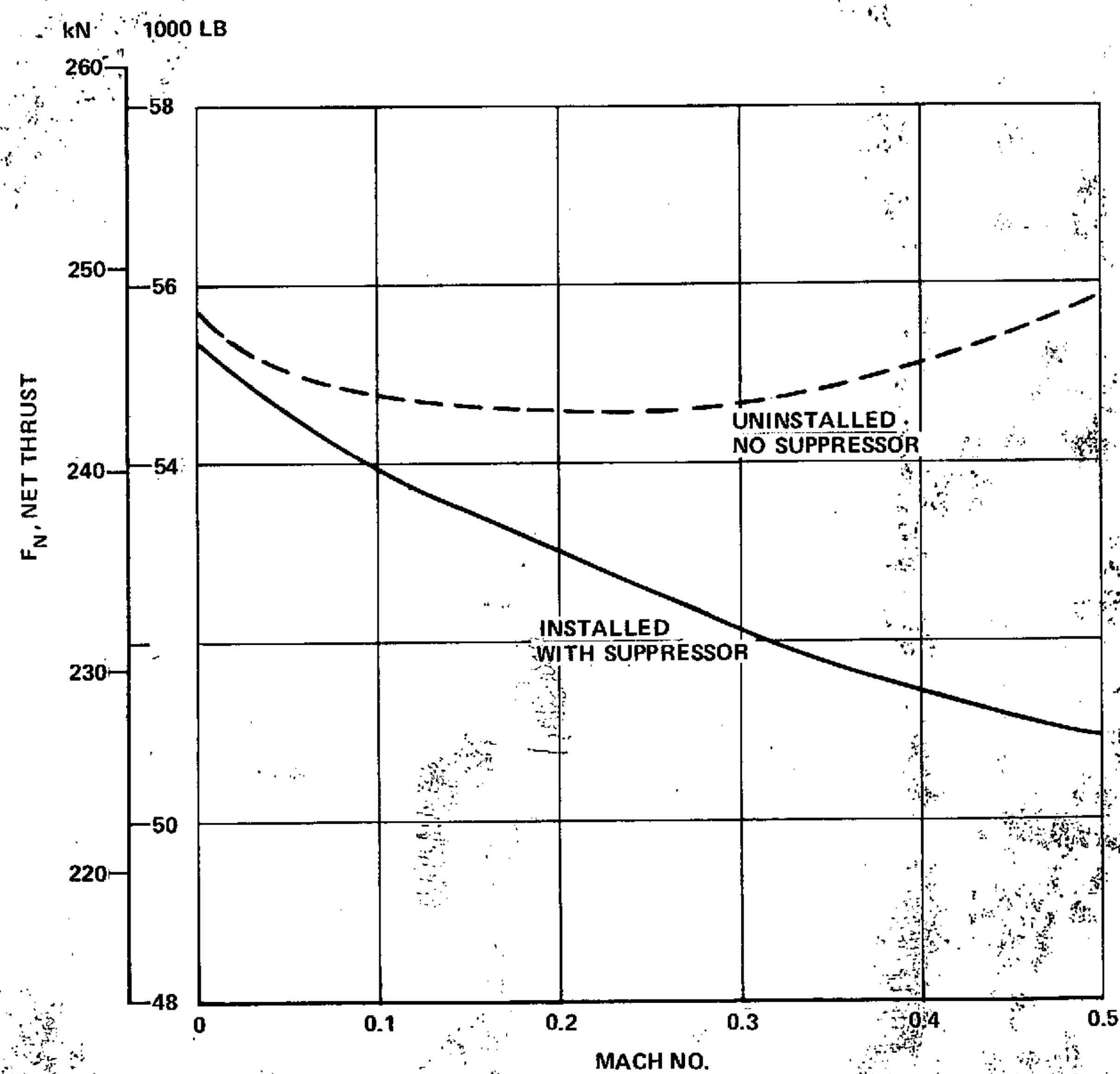


FIGURE 2-15. TAKEOFF PERFORMANCE

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GE MINI-BYPASS
STD DAY
WAT2 = 782 LB/SEC (354.6 kg/SEC)

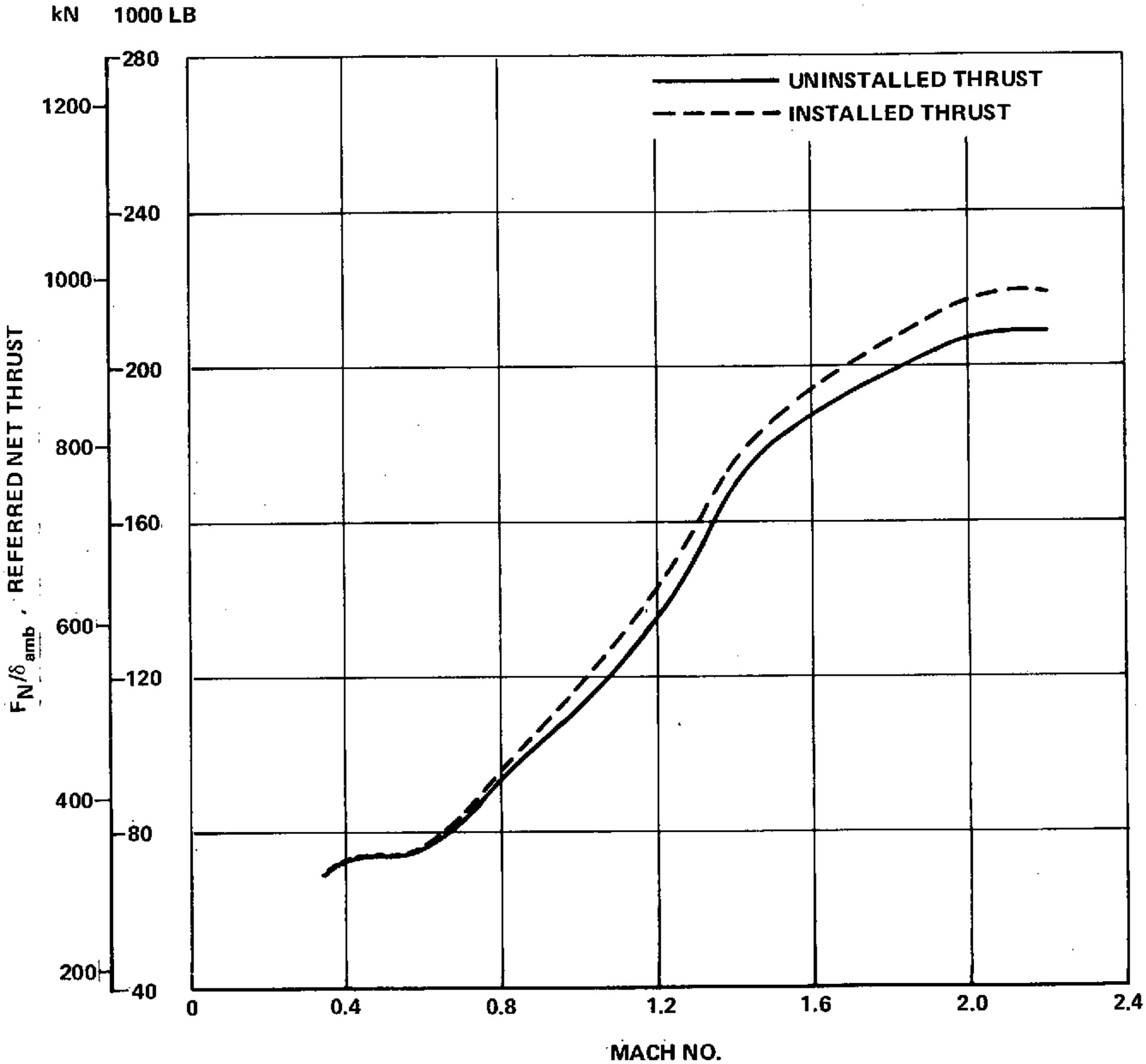

FIGURE 2-16. CLIMB THRUST

GE MINI-BYPASS
STD DAY
WAT2 = 782 LB/SEC (354.6 kg/SEC)

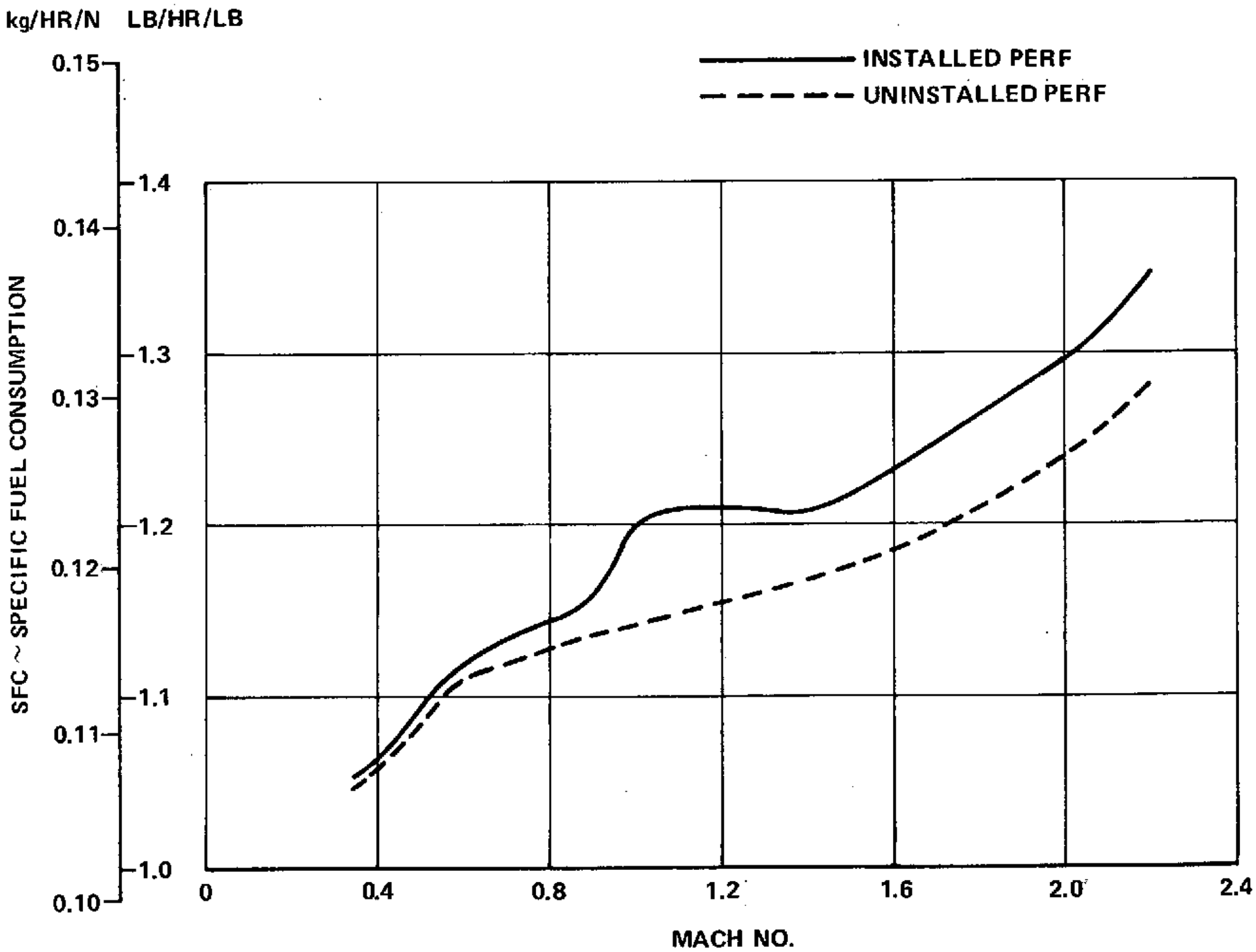

FIGURE 2-17. CLIMB SFC

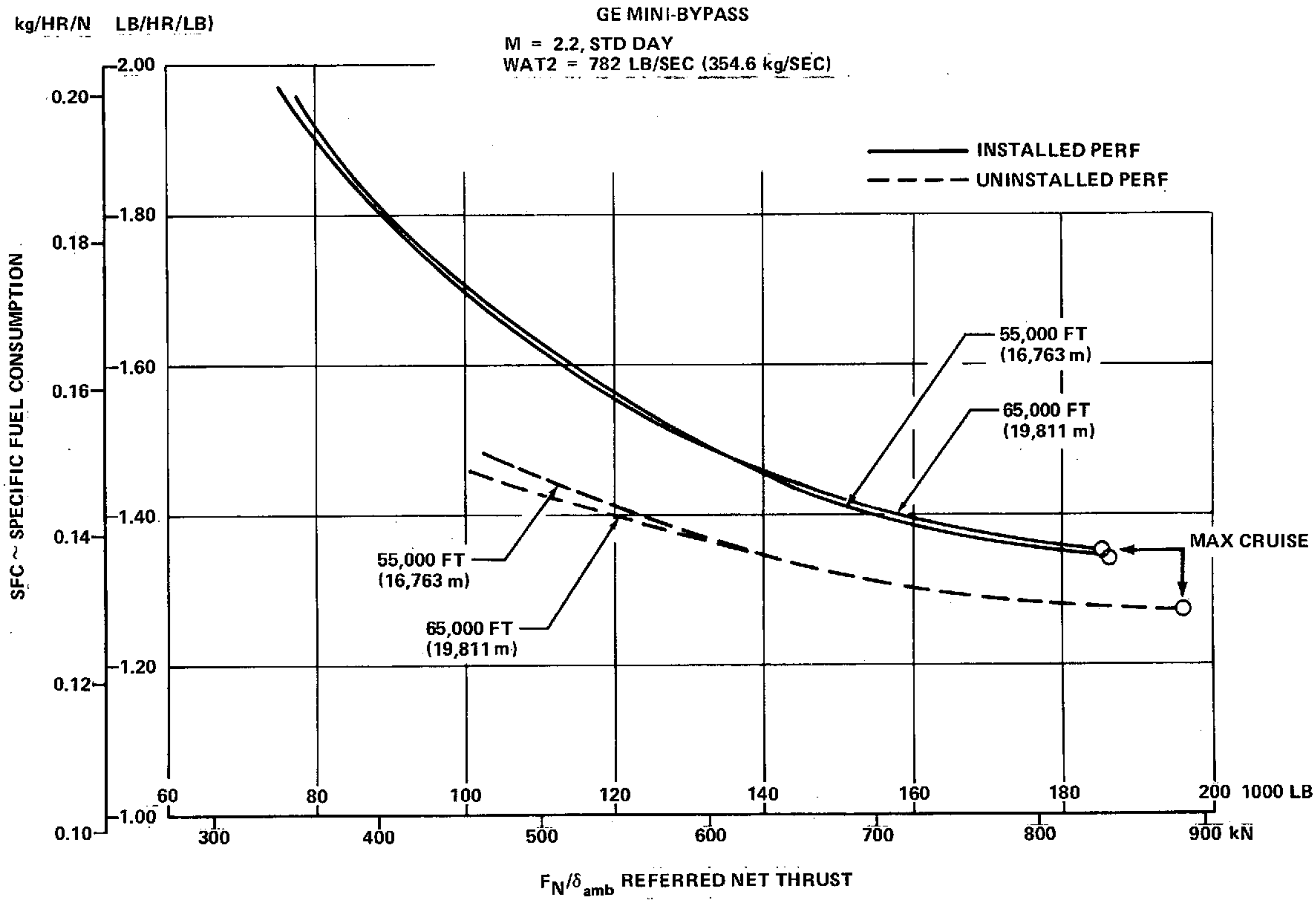

FIGURE 2-18. SUPERSONIC CRUISE PERFORMANCE

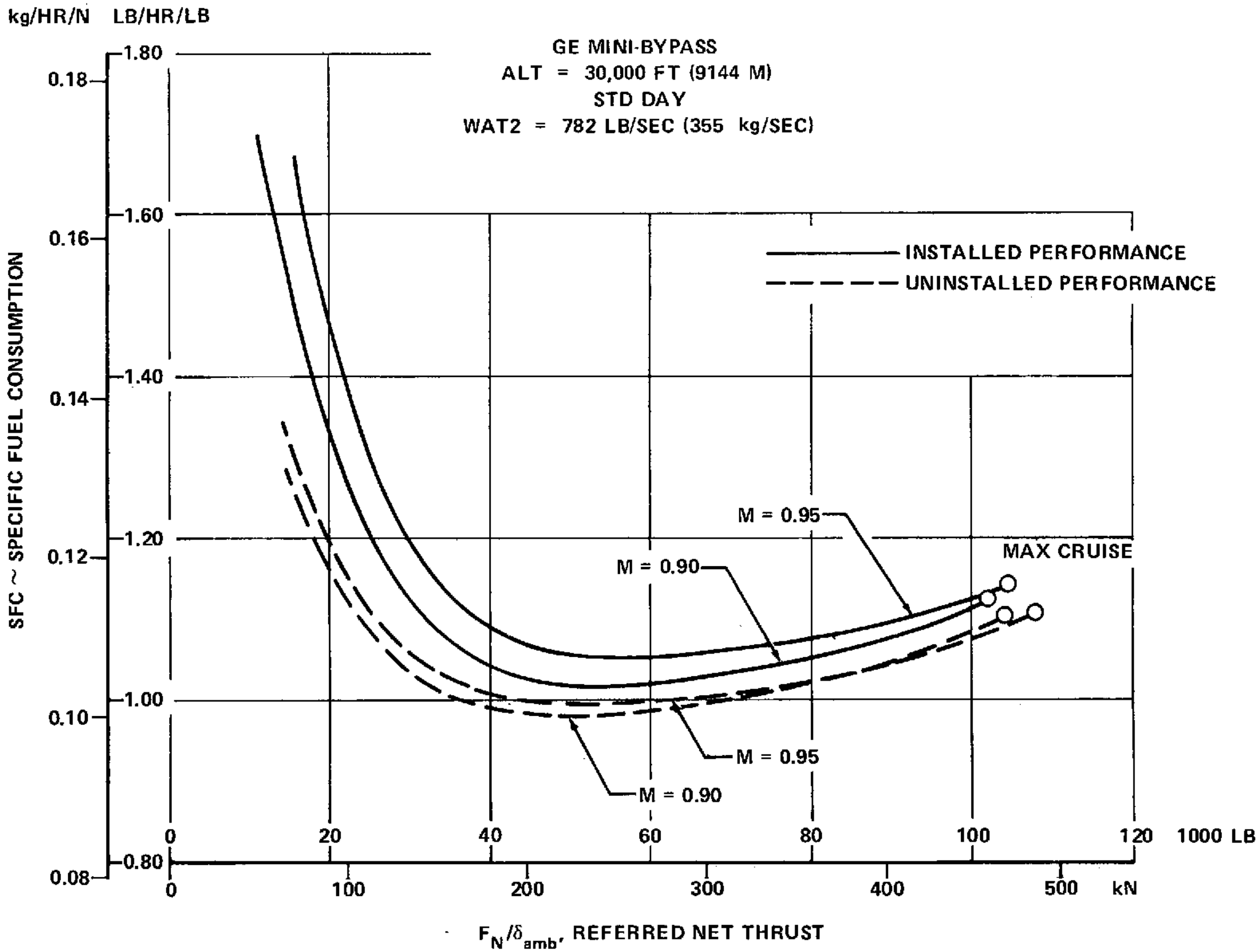

FIGURE 2-19. SUBSONIC CRUISE PERFORMANCE

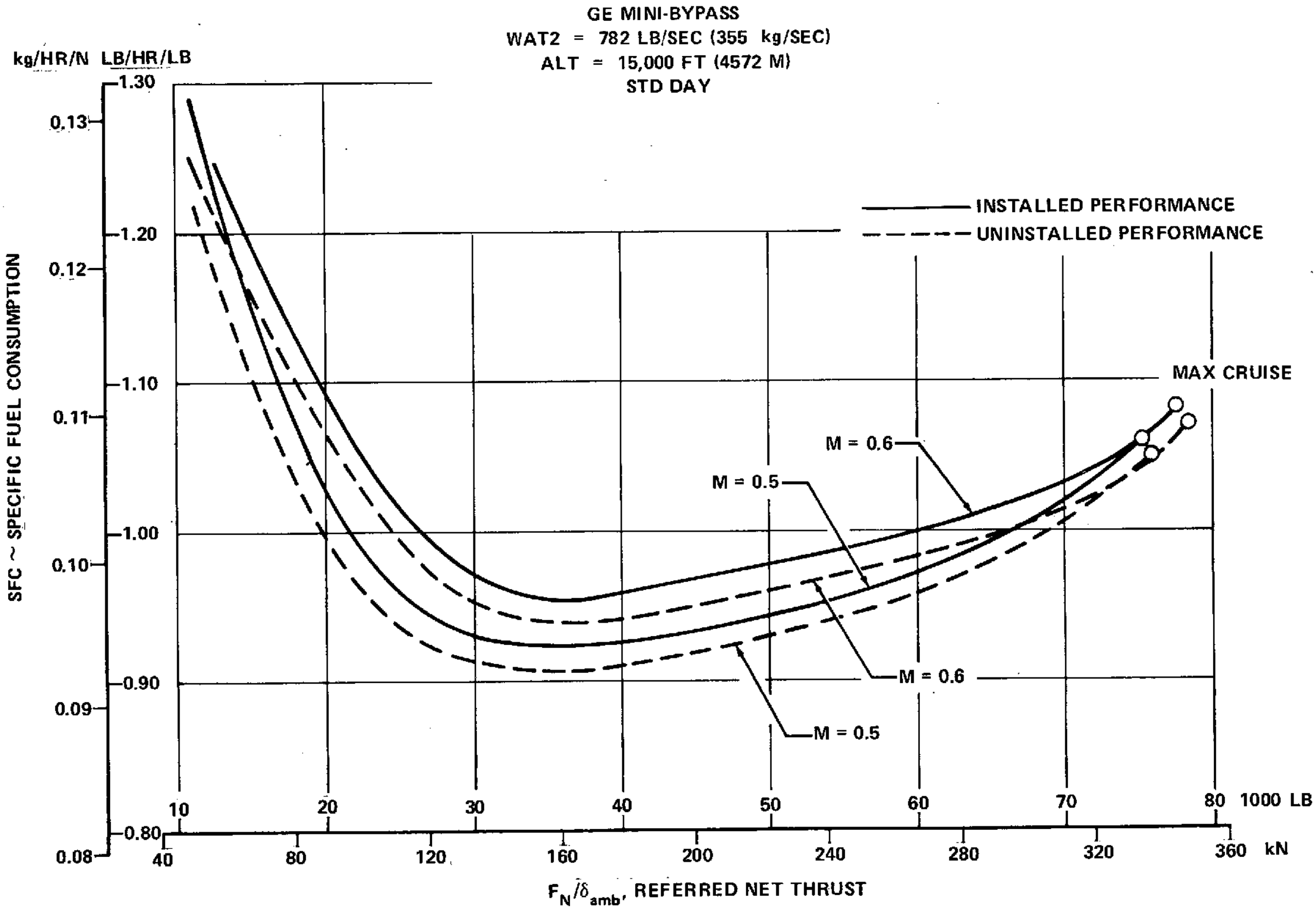

FIGURE 2-20. LOITER PERFORMANCE

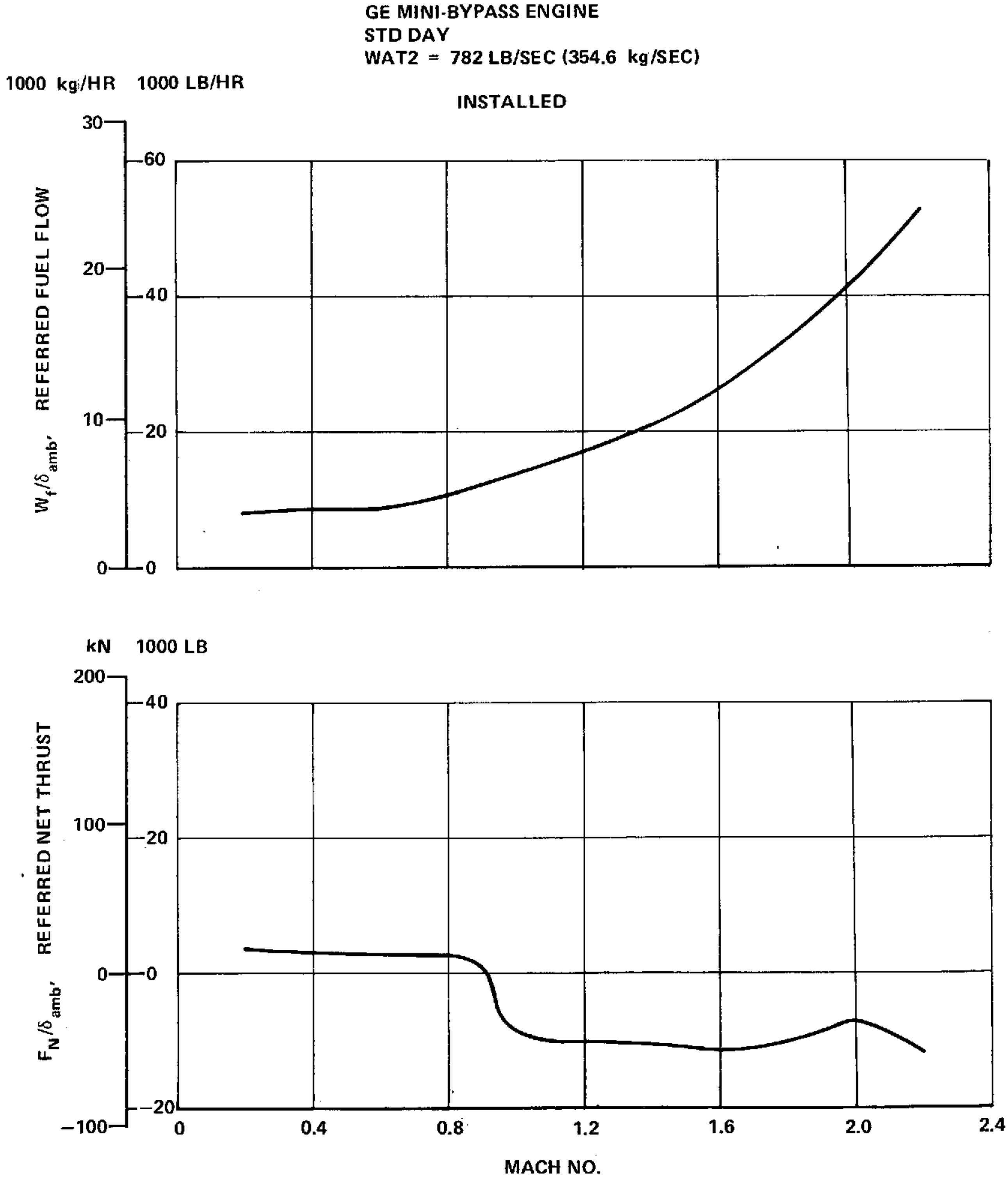

FIGURE 2-21. IDLE PERFORMANCE

Engine/Nacelle Location

Installation studies of the P7 engines in the baseline airframe in four axisymmetric nacelles have been completed. Inboard and outboard spanwise locations are the same as the -5A to maintain the existing wing torque box structure, disposition of control surfaces, and overall area distribution equivalent to the -5A baseline configuration.

The locations of the engines fore and aft are dictated by consideration of the following factors: aerodynamics requirement for intake face and maximum nacelle diameter location; power plant reverse thrust provision and airframe compatibility, and minimization of structural aeroelastic/flutter penalty while maintaining adequate provisions for the engine support pylons.

Due to aircraft control surface locations in relation to reverser nozzles, thrust can only be diverted in local areas (70°) above the upper wing surface and (150°) between the deployed flaps. (Figure 2-22.)

The engine locations, as shown on the three-view of the -5B configuration, (Figure 2-23) satisfy the afore mentioned criteria.

Engine/Nacelle Attachment to Wing

Engine mounting to the wing is by a three point attachment. The aft mount is on a box beam pylon cantilevered aft of the rear spar, and the two forward mounts are attached to structure provided on the rear spar.

The forward right hand mount carries thrust loads, vertical loads and side loads to the aircraft structure. The forward left hand mount transmits thrust and vertical loads only. The rear engine mount carries vertical loads and translates for engine expansion under operating temperatures. (Figure 2-22.)

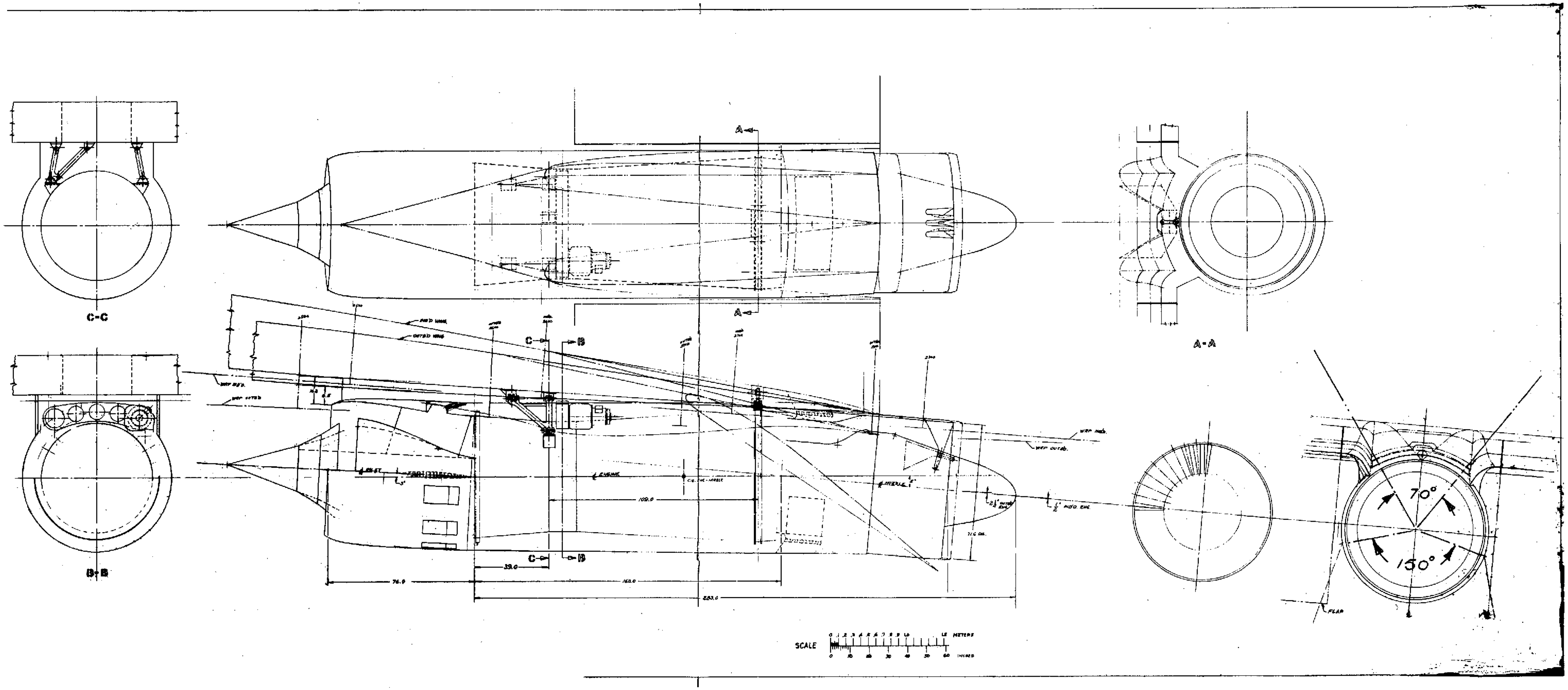

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FIGURE 2-22. ENGINE INSTALLATION SCHEMATIC

FOLDOUT FRAME 1

FOLDOUT FRAME 2

CHARACTERISTICS			
ITEM	WING	HORIZONTAL STABILIZER	VERTICAL STABILIZER
AREA SQ.FT.	10000	780	700
ASPECT RATIO	1.84	2.0	0.9
TAPER RATIO	.128 (TRAP)	0.16	0.25
SWEEP OF L.E.	71° & 57°	50°	50°
DIHEDRAL	0 AT T.E.	0	~
T/C %C	2.25-3.0	3.5	3.6
TAIL VOL COEF	~	.115	.044

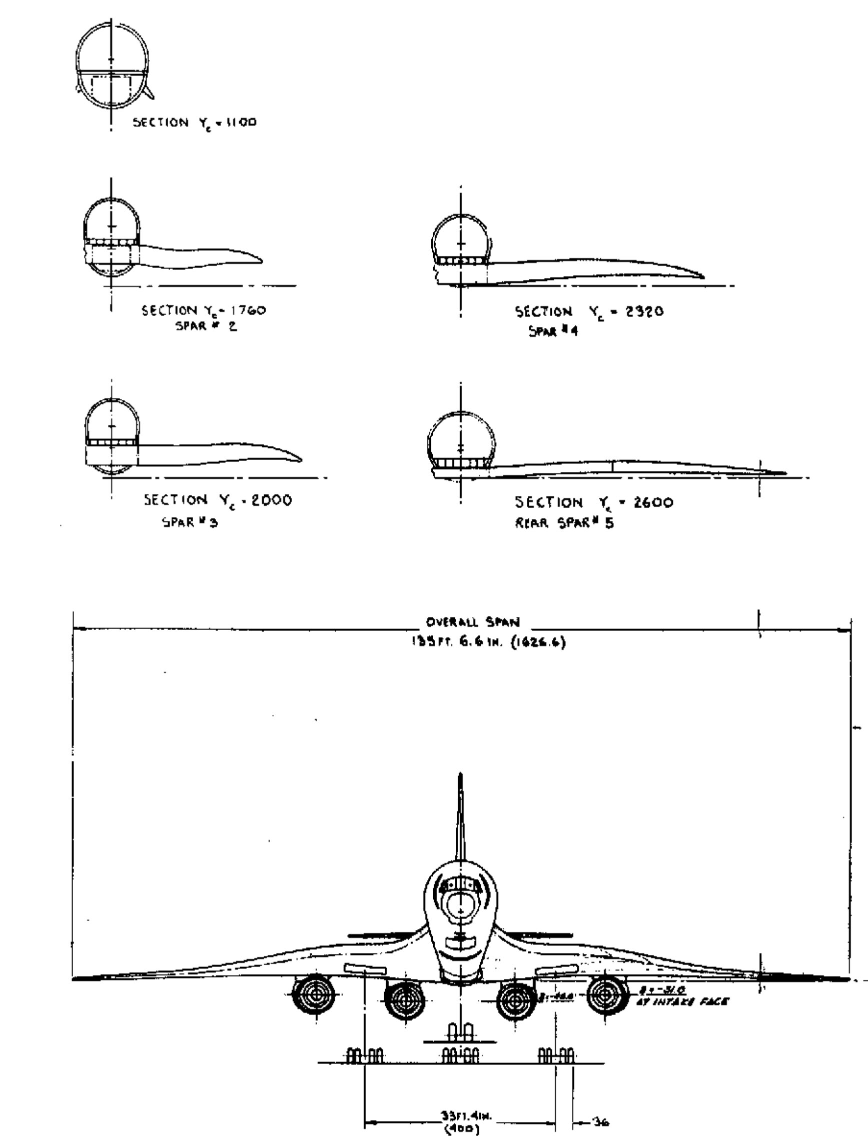
PAYLOAD CAPACITY

MIXED CLASS

FIRST CLASS - 3 & 4 ABREAST - 38 PITCH - 40 SEATS
TOURIST CLASS - 4 5 & 6 ABREAST - 34 PITCH - 235 SEATS
TOTAL 275 SEATS

CARGO VOLUME

FORWARD	1500 CU.FT.
AFT	750 CU FT
TOTAL	2250 CU FT



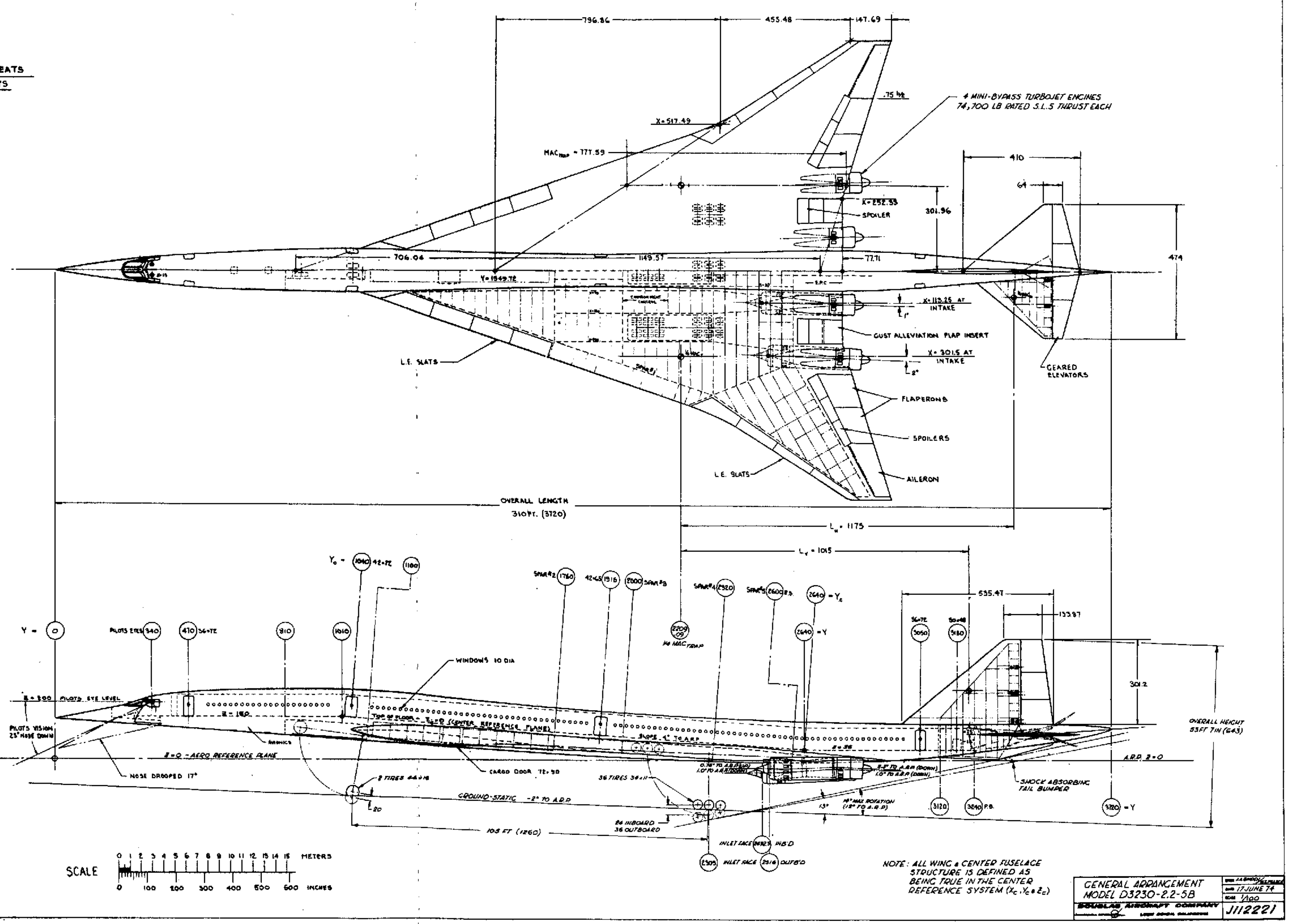


FIGURE 2-23. AST GE MINI-BYPASS ENGINE CONFIGURATION

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FOLDOUT FRAME 1

FOLDOUT FRAME 2

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Axisymmetric inlets are attached to the engine casing and are isolated from the wing structure to prevent transmission of wing deflection loads. This avoids distortion of intake geometry and loading of engine casing.

The boundary layer diverter is integrated into the engine nacelle/wing fairing.

Due to the decrease in weight of the P7 engine and an engine pod c.g. shift forward as compared to the baseline turbojet, the weight savings in engine support and airframe structure are considerable. (See weight statement.)

Other Configuration Changes

The reduced length of pod associated with the installation of the P7 engine eliminated the engine nozzle/ground clearance problem at maximum rotation that existed with the -5A baseline design.

Increase in ground clearance with the shorter pod, enabled the landing gear main and nose wheel struts to be shortened by 14 inches (35.6 cm). The associated structural benefits to this are shorter landing gear doors and a reduction in size and weight of the tail bumper and fairing. Minimum clearance of ground to rear fuselage on 14 degrees maximum rotation for the -5B configuration is 15 inches (38.1 cm).

Noise Estimates with Untreated Engines

In contrast to the use of the DAC designed integrated ejector/suppressor exhaust system for the -5A baseline airplane, it was decided that the engine manufacturers' sound suppressor and suppression data would be used for the engine airframe integration study.

The GE P7 engine selected for integration analysis incorporates a plug nozzle and a multi-element (chute type) suppressor with no ejector. GE supplied the basic engine cycle data, suppressed jet noise estimates, and physical characteristics for the jet noise suppressor.

Since the GE engine jet noise estimates represent suppressed levels, DAC elected to calculate the unsuppressed airplane noise levels first and then apply the GE estimated suppression characteristics to determine the suppressed noise level for use in sizing the engine (airplane). No attempt is made to analyze or verify the GE engine jet noise estimates. GE concurs in this approach.

The engine jet noise suppression characteristics applicable to flight were supplied by GE as shown in Figure 2-24. These data are reported to be based on static test results. GE assumed equal suppression in flight to that obtained statically, reportedly based on results of a NASA flight test program. It can be observed that GE lowered their "1973 goals" to a peak suppression level of 15 PNdB in flight from the "1972 goals" of 18 PNdB peak suppression. As this engine sizing study was nearing completion, information was received from GE stating that the GE suppression goals for the plug nozzle have been further reduced to approximately 10 PNdB in flight with no change in thrust loss characteristics.

GE CAPABILITIES FOR AST ENGINE CONFIGURATIONS
PEAK PNL JET SUPPRESSION RELATIVE TO CONICAL NOZZLE
- DESIGN POINT SUPPRESSION
- APPLICABLE TO FLIGHT

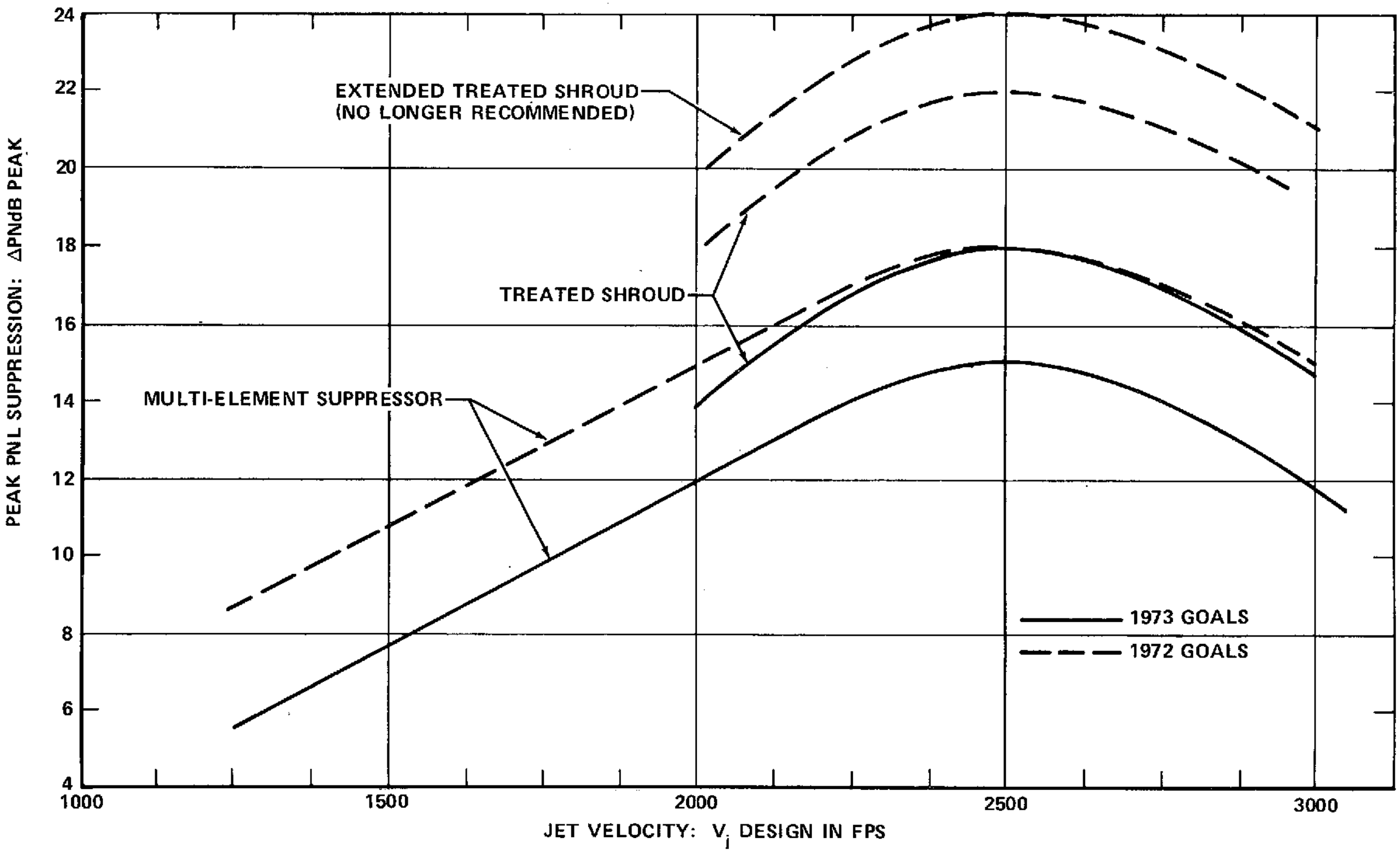


FIGURE 2-24. SOUND SUPPRESSION FOR GE MINI-BYPASS ENGINE

The unsuppressed jet noise levels estimated by Douglas for the airplane, with GE P7 engines, are as follows: [Engine flow rate = 782 lb/sec (355 kg/sec)].

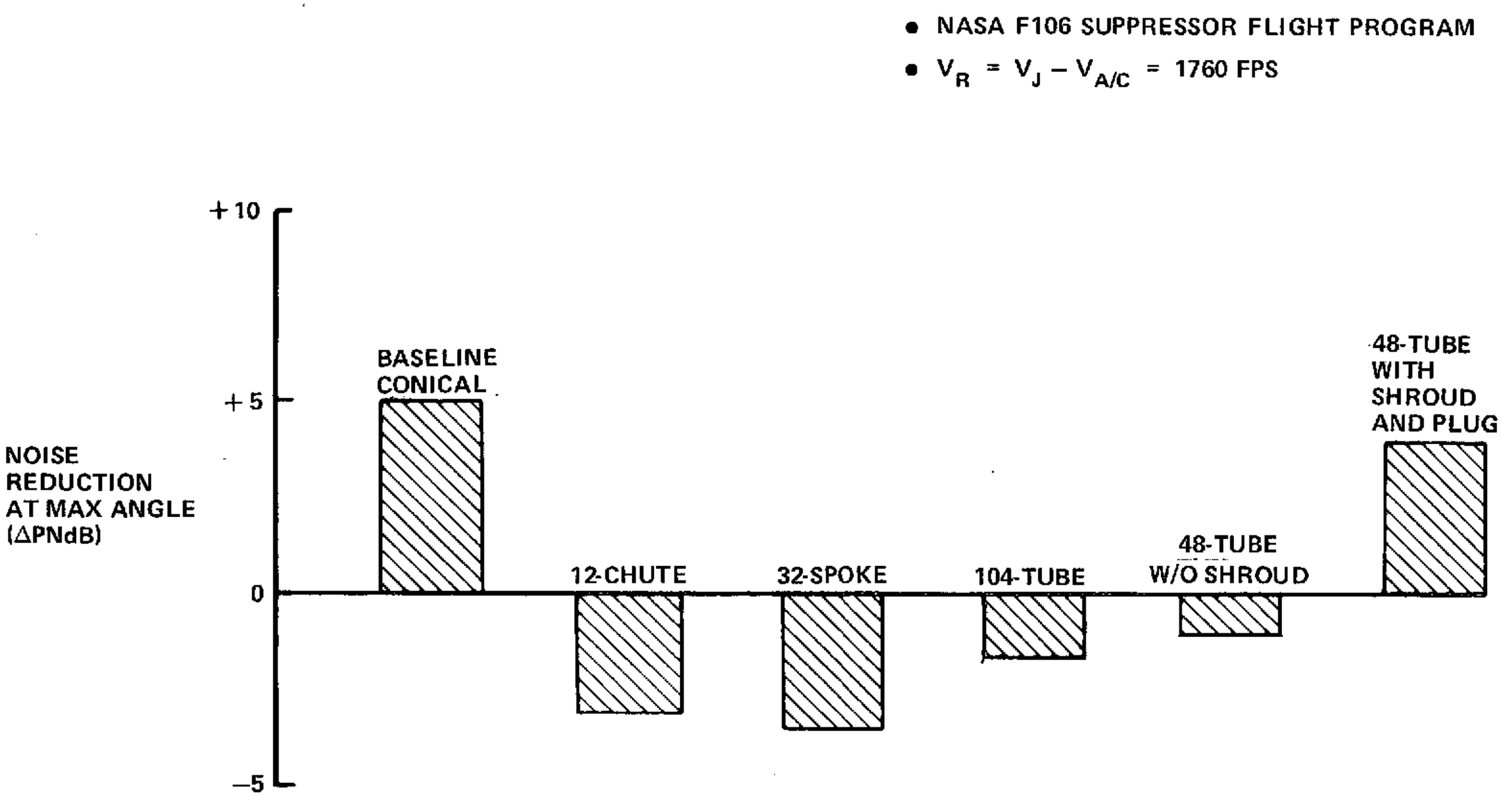
FAR PART 36 MEASURING STATION	DISTANCE, FT.(m)	UNSUPPRESSED TOTAL NOISE EPNL, EPNdB
Sideline	2270 (slant) (747)	119.8
Takeoff/Cutback	1290 (394)	119.5

Selection of the engine size using the GE "1973 goals" for suppression is described in detail in the engine sizing section.

Flight Effects on Noise Levels

The effects of forward motion on jet noise levels may be significant in estimating the amount of jet noise suppression obtained in flight. The jet noise levels measured in flight are functions of the relative jet velocity, the aircraft altitude, the atmospheric conditions and the type of exhaust nozzle. The flight effects for unsuppressed nozzles are described before the suppressed nozzles are discussed.

For unsuppressed nozzles, the influence of flight on measured jet noise levels varies with the type of exhaust nozzle installed. The most common type of exhaust nozzle in aircraft today is the plain circular nozzle. It has been well established that the noise level of a circular nozzle measured in flight is less than that measured in static tests for the same jet exhaust velocity. For example, data taken at constant jet exhaust velocity with conical nozzles on a DC-9 aircraft show significantly lower noise levels ($\sim$ 5 PNdB) in flight compared to the static level. Also, NASA flight data as shown in Figure 2-25 indicates a similar reduction in noise level for a conical nozzle. For a plug-type nozzle, the jet noise level in flight also decreases, but



SOURCE: NASA TMX-71439 AND DOUGLAS DATA

FIGURE 2-25. FLIGHT EFFECTS OF CONICAL BASELINE AND SUPPRESSOR NOZZLES apparently at a lower rate than the equivalent conical nozzle when both nozzles are compared at constant jet exhaust velocity. This tendency is also shown in some of the NASA flight test results.[1] The precise reasons for the different response of the plug nozzles in flight are not well understood at this time.

In the case of jet noise suppressors, the noise levels measured in flight are often less than static levels, but the reductions are usually not as large as for unsuppressed conical nozzles. The inherent tendency of jet noise suppressors to shift some of the noise to higher frequencies may account for this increase in noise. NASA flight test results show a definite trend of the high frequencies in the spectrum to exhibit higher sound pressure levels (SPL) in flight than statically[2], [3]. The reasons for the increase in noise of the high frequency portion of the spectrum due to forward speed are not well understood at this time. Figure 2-25 shows the relative performance of 5 suppressors tested in the NASA program. It can be observed that all 4 test suppressors without shrouds experienced an increase in noise level in flight. The average value of this increase is approximately 2.3 PNdB. One suppressor with a shroud showed a noise reduction with forward speed. It is believed that properly designed suppressors can achieve flight performance equal to, or better than, the baseline conical nozzle.

The latest information from GE indicates a revision of the GE 1974 suppressor status to 12 PNdB peak static suppression and 10 EPNdB peak suppression in

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(1) Burley, R. R. and Karabinus, R. J., "Flyover and Static Tests to Investigate External Flow Effect on Jet Noise for Non-Suppressor and Suppressor Exhaust Nozzles", NASA TXM-68161, January, 1973.

(2) Burley, R. R. and Johns, A. L., "Flight Velocity Effects on Jet Noise of Several Variations of a Twelve-Chute Suppressor Installed on a Plug Nozzle", NASA TXM-2918, January, 1974.

(3) Burley, R. R. and Head, V. L., "Flight Velocity Effects on Jet Noise of Several Variations of a 48-Tube Suppressor Installed on a Plug Nozzle", NASA TMX-2919, January, 1974.

flight as shown in Figure 2-26. GE has set the "1985 goal" as 12 EPNdB peak suppression in flight relative to a circular nozzle and a gross thrust coefficient greater than 0.91. From the NASA GE test data DAC has reviewed, it appears that realistic values of in flight suppression would be lower than the GE 1974 estimated status; however, the "1985 goal" could be realistic providing an effective suppressor configuration is developed for in flight operation.

These lowered suppression goals were significant enough to cause GE to recommend another engine cycle, the "double-bypass dual cycle" engine in the GE ninth monthly report to NASA, September, 1974.

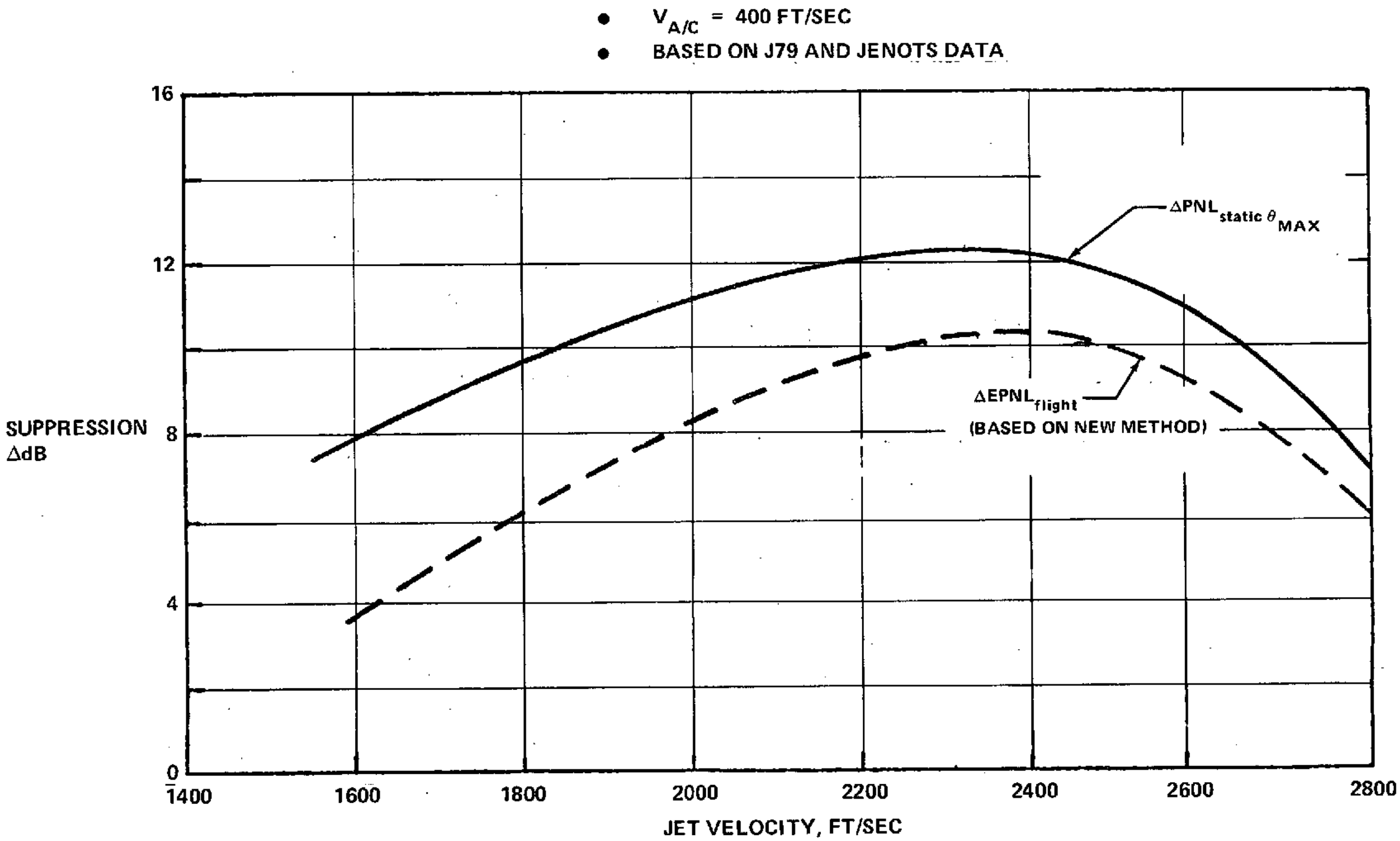
The initial indication that GE was revising the peak suppression goal to a level of 10 PNdB or less, led DAC to the consideration of an alternate DAC exhaust nozzle configuration which Douglas believes can realize higher levels of suppression. The nozzle selected is the DAC integrated ejector suppressor system identical to that incorporated in the baseline configuration -5A.

Estimates with the Douglas Integrated Ejector Suppressor Exhaust System

The estimated jet noise suppression levels in a flight environment for the baseline configuration exhaust system are as follows:

FAR Part Measuring Station	Total Noise Suppression
Sideline	11.9 EPNdB
Takeoff/Cutback	10.4 EPNdB

These estimates are based on a well ventilated ejector nozzle utilizing empirical loss coefficient correlations. It is imperative that complete, or nearly complete, mixing be attained in practice to achieve maximum noise



SOURCE: GE REVIEW
NASA LEWIS 10/22/74

FIGURE 2-26. STATIC AND FLIGHT JET NOISE CHARACTERISTICS OF 32 CHUTE SUPPRESSOR reduction. Tests are underway at present to verify this mixing efficiency.

Preliminary design methods are applied to estimate the exhaust system suppression utilizing DAC calculated unsuppressed jet noise levels based on installed engine data and the above estimated jet noise peak suppression levels. The results indicate that the GE mini-bypass engine with the Douglas integrated exhaust system (see Figure 1-4) can be sized to meet FAR Part 36 requirements at 773 lb/sec (351 kg/sec) inlet corrected airflow, as discussed in the supplemental paragraph of this section.

STRUCTURAL ANALYSIS

Structural Model

The baseline -5A structural model used in these engine-airframe studies is shown in Figure 2-27. The model has been developed using an interactive computer graphics program which enables the user to generate geometric and structural data quickly and accurately. The program also enables data for plain or numbered diagrams to be output for subsequent processing by a Gerber plotter. The model consists of a half-fuselage from station y = 1100 (nose gear attachment) to y = 3505 (rear spar of horizontal tail). This extensive idealization also enables accurate stiffness data to be generated for aeroelastic and flutter analyses. The model contains 1283 bars, 1008 panels, 3574 stresses, 3938 element forces, 3647 degrees of freedom with five applied load conditions, and three fully stressed design iterations.

Structural Optimization

The current optimization capability for statically loaded structures consists of a resizing subroutine, ARROW (Automated Redesign and Reanalyses for Optimum Weight) which is part of the FORMAT analysis system. The optimization procedure is as follows:

a. Basic structural data from the computer graphics program is input to the FORMAT Phase 1 module which generates as output, matrix data required by the subsequent analysis and resizing sequence.

b. FORMAT Phase 2 executes user-defined matrix instructions to analyze the structure. Initial element sizes are available from previous studies used in the development of the baseline configuration.

c. Following the analysis, the ARROW routine is exercised. The user may specify a resizing option for defined stress and/or stiffness constraints. The modified sizes are returned by ARROW for a further analysis. The process continues for a number of iterations (usually between 2 and 6)

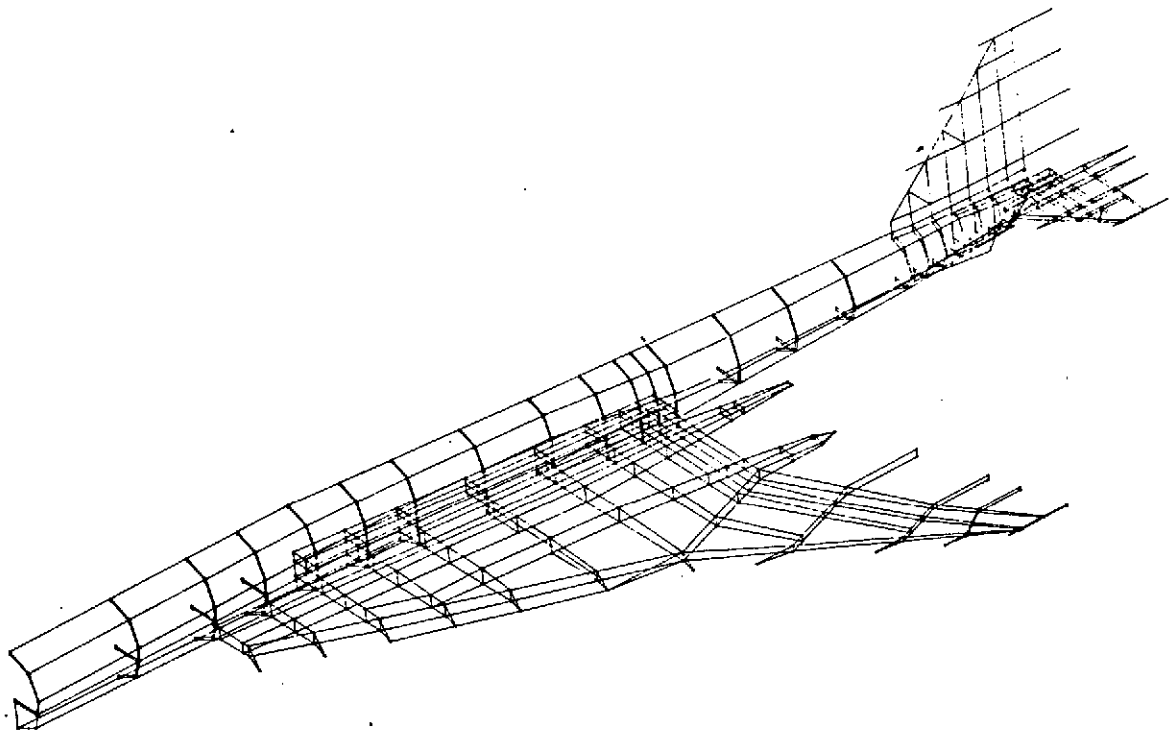


FIGURE 2-27. AST STRUCTURAL ANALYSIS MODEL – 5A

specified by the user, or until ARROW detects that a minimum weight structure has been achieved. For this study, each configuration examined was resized twice to satisfy specified strength allowables. Structural influence coefficients (SIC's) are calculated for each final design. Two reduced sets of load vectors are specified for aeroelastic and flutter calculations.

d. Applied Loads

(1) Aerodynamic Loads

Quasi-static aeroelastic loads and deflections are calculated, using aerodynamic and structural influence coefficients, with the Matrix Aeroelastic Loads System (MALS) Program. MALS is currently operational as a module within the FORMAT system.

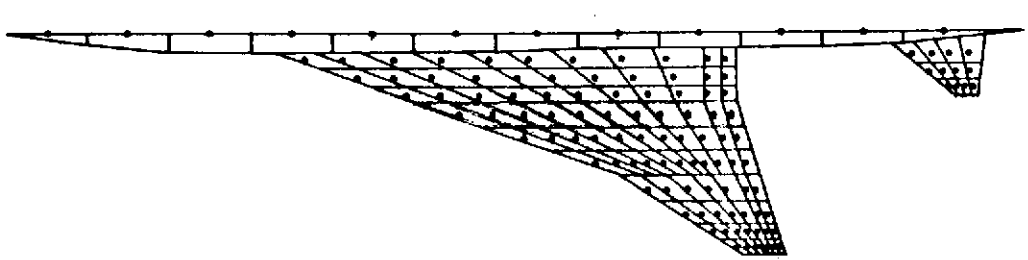
Aerodynamic influence coefficients (AIC's) are calculated using the Doublet Lattice Method, the Method of Images, and the Woodward Program. The aerodynamic idealization of the baseline configuration is illustrated in Figure 2-28.

Structural influence coefficients are generated, as noted previously, following the structural optimization. For initial loading estimates, before SIC's are known, a rigid structure is assumed.

Inertia force influence coefficients are included so that trimmed loading conditions may be obtained.

(2) Inertia Loads

Inertia loads for the baseline design are used to account for payload, structure and systems, fuel and landing gear. These are



133 POINTS
145 DEGREES OF FREEDOM

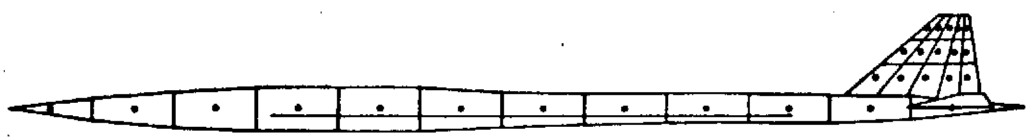
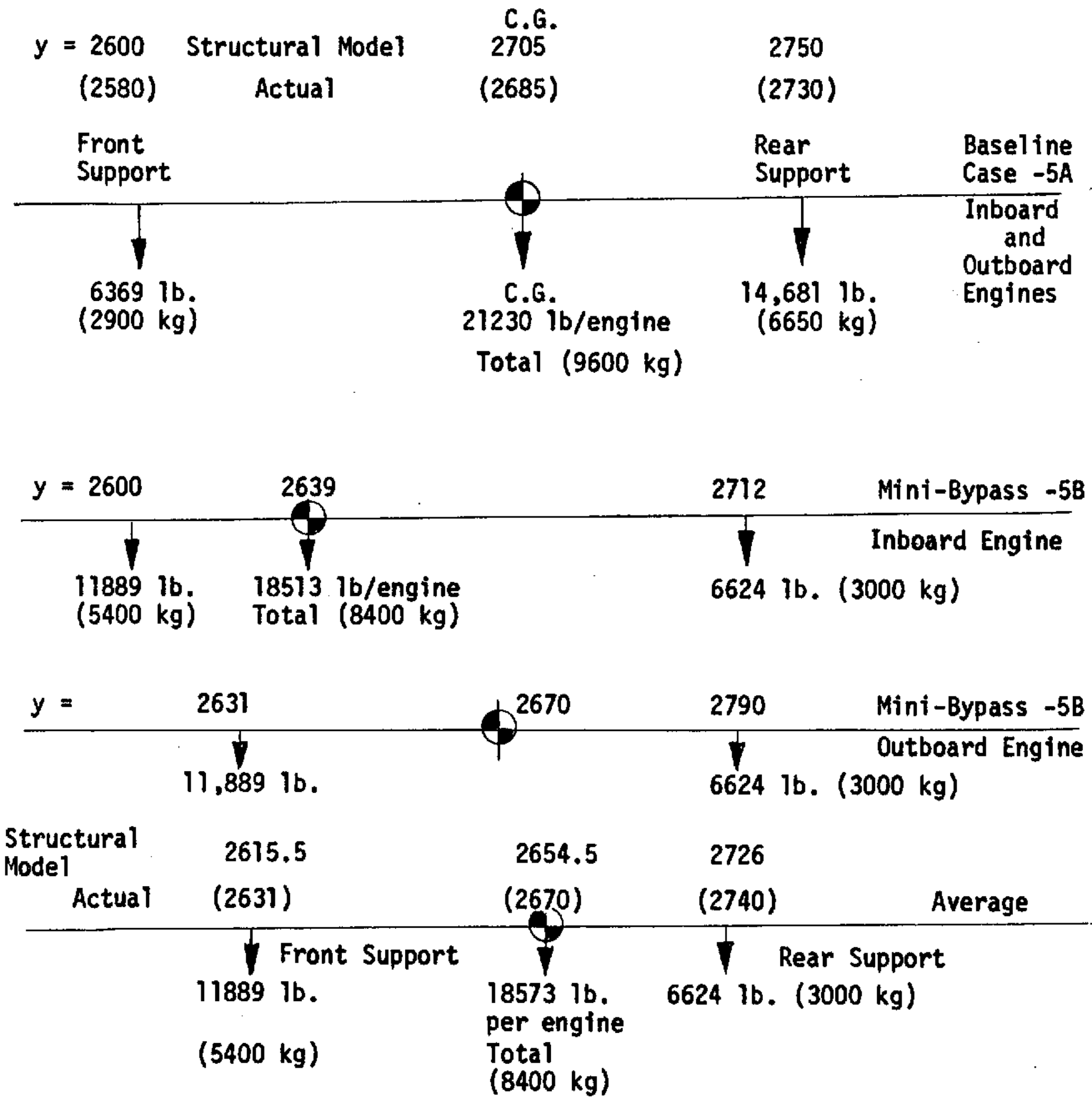


FIGURE 2-28. STRUCTURAL ANALYSIS AERODYNAMIC LOADS MODEL – 5A

assumed unaffected by the engine configuration changes in this study. Fuselage pressure loadings are included.

Engine loads and locations for the configurations considered are summarized below. Weights shown are for one quarter of the total propulsion system, i.e., 1g weight per engine installation.



e. Applied Load Conditions

The following loading conditions are applied to all configurations for strength sizing:

Condition 1: Subsonic 2 1/2 g symmetric maneuver

Condition 2: Supersonic 2 1/2 g symmetric maneuver

Condition 3: Supersonic -1 g symmetric maneuver

Condition 4: 4 g vertical

Structural Analysis

The optimization of the -5B P7 engine airplane design gave an idealized model weight of 38,310 lbs. (17,377 kg) per half airplane. This value is then modified to account for recalculated trimmed tail-loads. The change in applied bending moment is plotted in Figure 2-29 where it can be seen that the main effect of trim is on the loads at the rear fuselage. The internal bending moment at a representative rear fuselage station is calculated, and the bending moment correction (Figure 2-29) represents an average 7-1/2 percent increase. The increase in fuselage longeron sizes necessary to carry this increased moment is calculated as 138 lbs. (63 kg). The revised -5B model weight is, therefore, 38310 + 138 = 38,448 lbs. (17,440 kg). This represents a decrease of 1007 lb/side (457 kg) or approximately 2-1/2 percent less than the -5A baseline structural model.

The newly optimized design is examined in detail to determine the contribution of various structural components to the weight difference, and to validate that size changes are consistant with changes in loading conditions. The results, with 78 percent of the weight changes analyzed, are summarized in Table 2-4. The following conclusions are made:

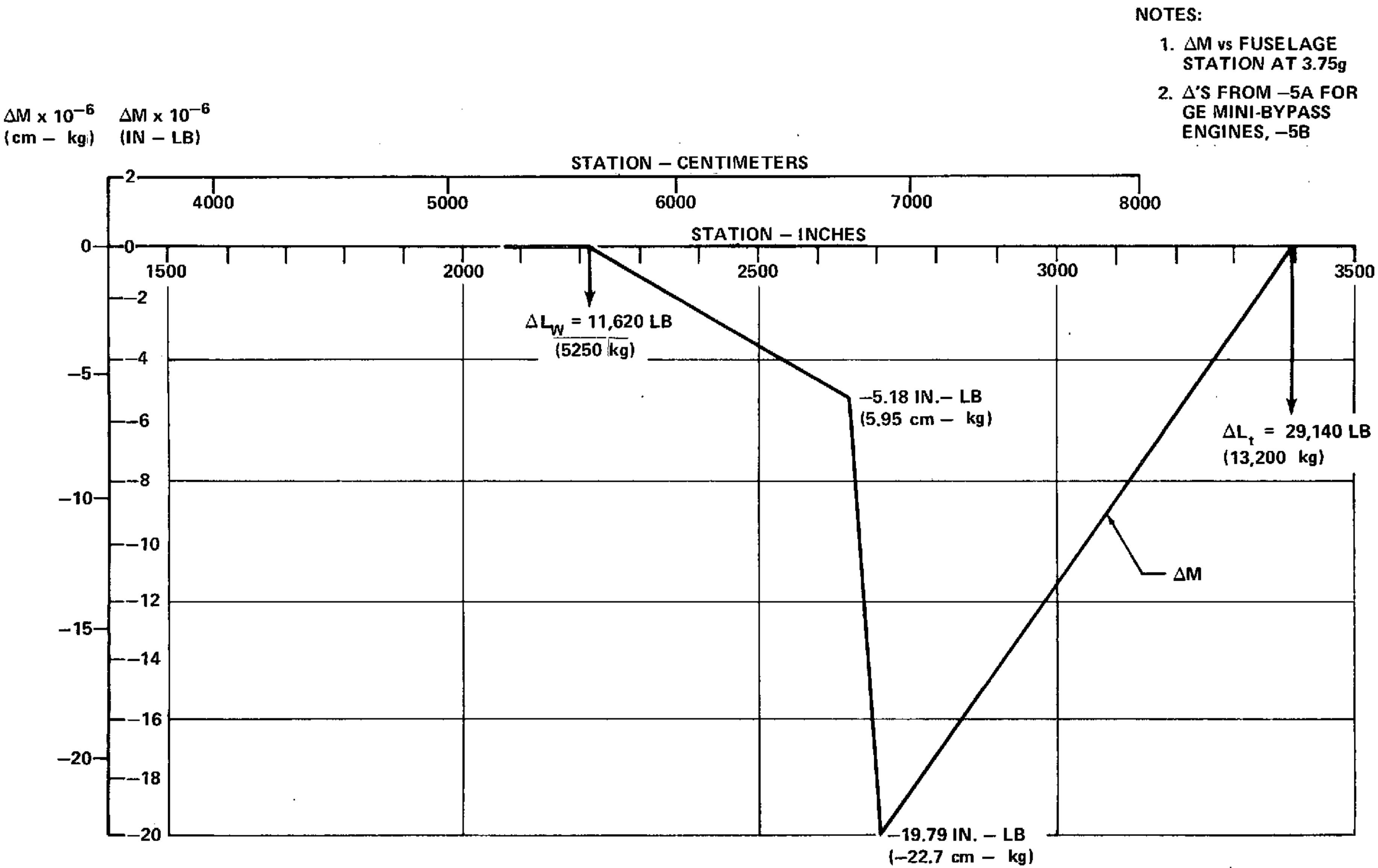


FIGURE 2-29. ULTIMATE BENDING MOMENT DIAGRAM

a. Few significant changes in design loading conditions are noted when comparing the baseline and P7 configurations. Details are given in Table 2-4.

b. Forty-two percent of the weight decrease is found in the nacelles. Examples of actual size changes are shown in Figure 2-30.

c. Due to the reduced relieving moment from the lighter engine, the bending material in the region of the rear spar shows a small increase as expected. Actual size changes are shown in Figure 2-30.

d. Wing structural changes from about 80 in. (2.03 m) forward of the rear spar are negligible.

e. Fifty-eight percent of the weight decrease takes place in the fuselage and wing carry-through structure. This is due to reduced fuselage bending due to the lighter engine and is concentrated largely in the top and bottom fuselage longerons and side shear panels. Typical variations in longeron cross sectional areas are plotted in Figure 2-31.

f. Aeroelastic changes

(1) Aeroelastic changes in wing effectiveness ($\eta = \frac{C_{L_{\alpha_w}}^{E}}{C_{L_{\alpha_w}}^{R}}$) versus q and the ΔXa.c. (change in aerodynamic center location) versus q are shown for the -5A base and the mini-bypass engine configuration -5B with the structural weight removed for optimum strength; Figure 2-32. The expected change in wing effectiveness at maximum climb speed and M = 0.8 is considered to be negligible. The decrease in ΔXa.c. requires a slight increase in horizontal tail balancing load which is well within the aircraft capability.

(2) Analysis of the -5A design, Table 1-1, indicates that 2860 pounds (1300 kg) more than required for strength is required for the -5A baseline configuration to meet the aeroelastic and flutter require-

TABLE 2-4
MODEL WEIGHT COMPARISON -5A AND -5B
(POSITIVE WEIGHT DIFFERENCE INDICATES -5A HEAVIER THAN -5B)

AREA OF STRUCTURE	WEIGHT DIFFERENCE PER HALF AIRPLANE (-5A – -5B) LB	kg	DESIGN LOAD CONDITION(s) MINI-BYPASS IN ORDER OF IMPORTANCE	SIGNIFICANT CHANGES FROM BASELINE
INBOARD NACELLE	143.7	65.1	1, 4, 2	4, 1, 2
OUTBOARD NACELLE	279.8	127.0	4, 1, 2	4, 2, 1
INBOARD REAR SPAR	–98.8	–41.6	1, 2	NONE
SPAR NO. 4 (L.G. ATTACH)	NEGLIGIBLE		1	NONE
TOP SKIN, SPARS 4, 5	NEGLIGIBLE		4, 2	NONE
FUSELAGE IDEALIZED LONGERON, TOP ℄	127.6	57.8		
2ND LONGERON	53.6	24.3		
3RD + 4TH LONGERONS	ZERO (MIN GAGE)		1, 4	NONE
5TH LONGERON	103.8	47.1		
6TH LONGERON (BOTTOM ℄)	72.5	32.8		
TOP FUSELAGE, IDEALIZED SKIN PANEL	4.4	2.0		
2ND SKIN PANEL	18.0	8.2		
3RD SKIN PANEL	31.6	14.3	4, 1	NONE
4TH AND 5TH (BOTTOM) PANELS	36.0	16.3		
TOTAL	779.2	352.0		

NOTES:

1. TOTAL WEIGHT DIFFERENCE -5A -5B = 1007 LB (455 kg). BREAKDOWN ABOVE ACCOUNTS FOR 78 PERCENT. REMAINDER IS IN WING CARRY-THROUGH STRUCTURE.
2. WEIGHTS ABOVE INCLUDE 138 LB (62.5 kg) ALLOWANCE FOR MODIFIED -5B TAIL LOADS.
3. LOAD CONDITIONS: 1 = 2-1/2g SUBSONIC SYMMETRIC MANEUVER.
   2 = 2-1/2g SUPERSONIC.
   3 = –1g SUPERSONIC
   4 = 4g LANDING

KEY TO STRENGTH-OPTIMIZED BAR SIZES

X.X BASELINE –5A

X.X GE MINI-BYPASS, –5B

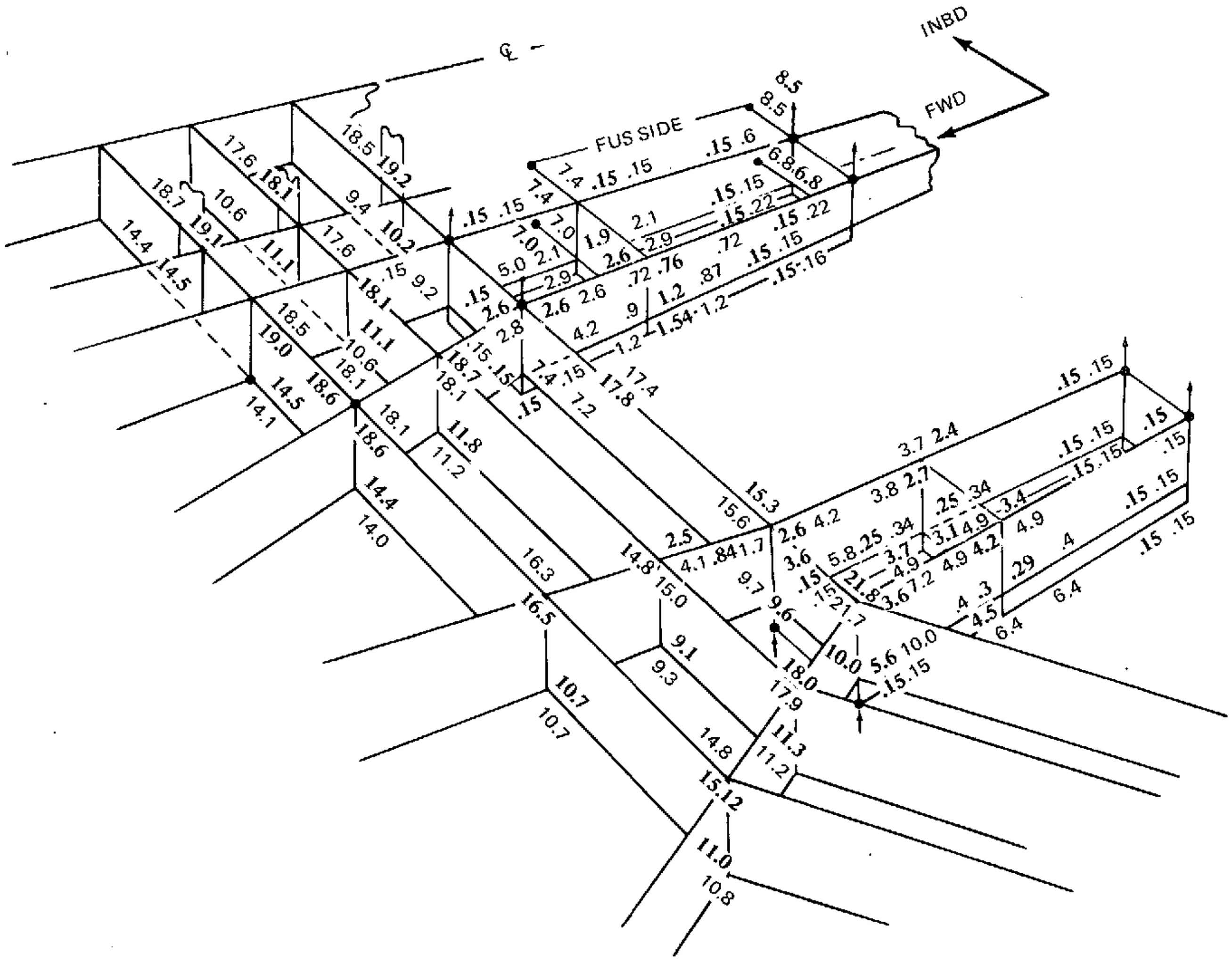


FIGURE 2-30. EXAMPLE OF STRUCTURAL ELEMENT ANALYSIS

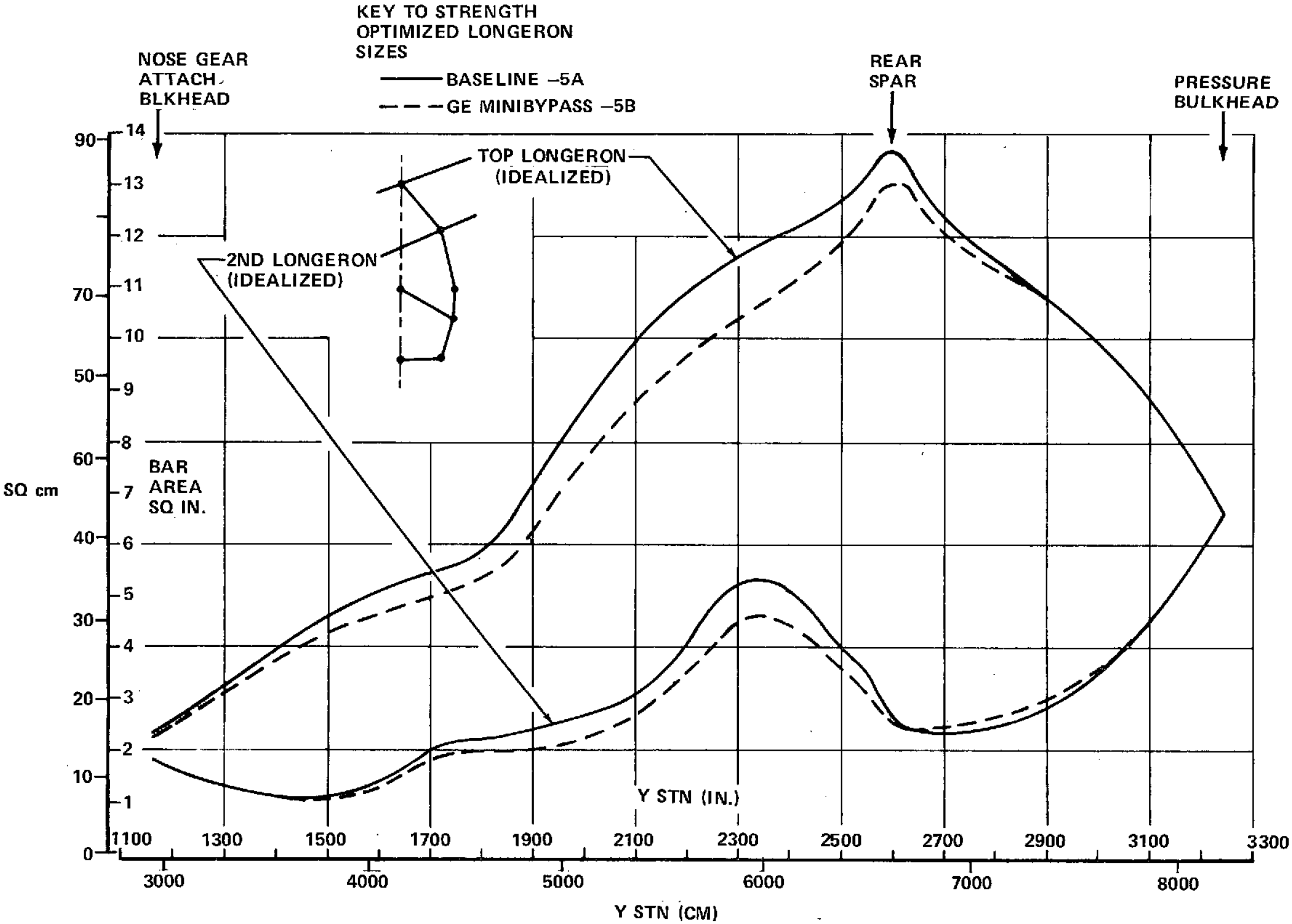


FIGURE 2-31. TYPICAL FUSELAGE LONGERON SIZE VARIATIONS

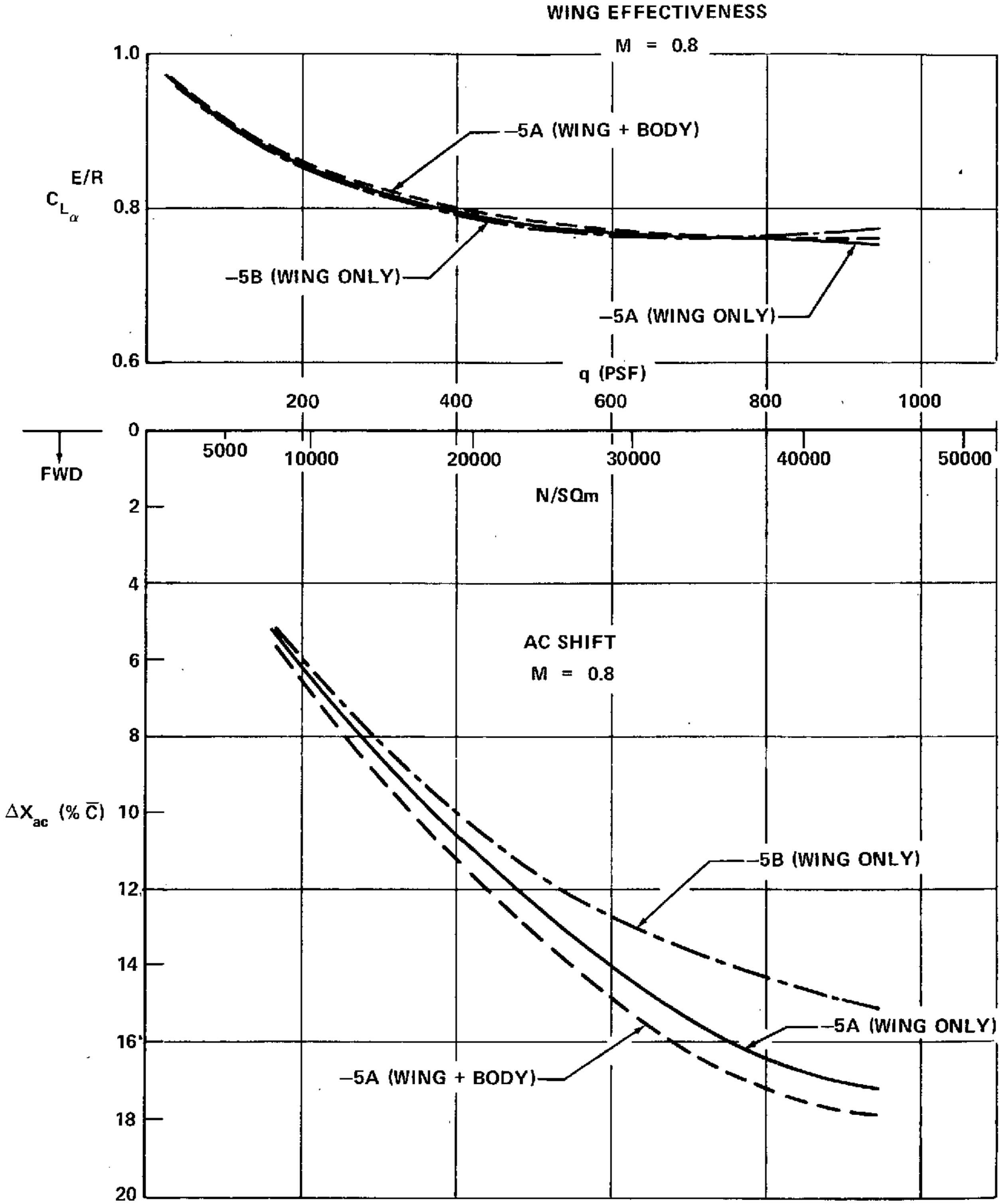


FIGURE 2-32. AEROELASTIC CHANGES TO WING AND WING-BODY

ments. Of this 2000 lb. (907 kg) is for aeroelasticity and 860 pounds (390 kg) is for flutter.

g. In order to facilitate estimates of structural weight changes from variations in engine weight or location, the idealized structural model for the -5A is modified for an 80 inch (2.03m) forward engine center of gravity location change and an 80 inch (2.03m) aft engine center of gravity change. Figure 4-27 shows the results of this study holding the baseline -5A engine size and weight constant. Point 1 of Figure 4-27 is the original base -5A structure, point 2 is the aft 80 inch (2.03m) movement, and point 3 the 80 inch (2.03m) forward movement.

h. The mini-bypass engine studied in the structural analysis model produces the weight increment shown in Figure 4-27 as point 4 and after correction for the trimmed balancing tail load results in a structural optimized weight savings of 2014 pounds (914 kg) per airplane, or 1007 pounds (457 kg) per side, shown as point 5 in Figure 4-27.

   This estimated structural weight change has been modified subsequently to a value of 1300 lb. (590 kg) savings, as indicated in Figure 4-27 and reflects the revised center of gravity location for the mini-bypass engine. This weight increment is included in the weight statement, Table 2-5.

Flutter Analysis

Additional structural optimization of the configuration described above provided the Structural Influence Coefficients for flutter analysis. The flutter analysis results are discussed in more detail in Section 4 under structural analysis. Briefly, the flutter speed is found to be primarily responsive to the engine weight and less responsive to small shifts in the location of the c.g. for the engines studied. Therefore, the two extreme weight conditions, the lightest being the duct heating turbofan engine configuration (-5C), and the heaviest, the variable cycle engine (-5D), were analyzed for flutter and compared to the turbojet (-5A) baseline case. An 860 pound (390 kg) structural weight flutter increase is predicted to be sufficient to meet the flutter requirements for both weight extremes (see Section 4). Therefore, since the P7 engine weight is between these two extremes, the 860 pound (390 kg) structural weight allocated for flutter should be adequate for the P7 engine configuration (-5B).

WEIGHT ANALYSIS

Section 1 provides a description of the methods used in defining detail weights for the baseline -5A. Weights in this section are calculated based on detail analysis of drawings where the airplane changes are defined to include the P7 engines.

Table 2-5 compares the weight of the airplane with mini-bypass engines (-5B) to the turbojet baseline (-5A). Weight differences are limited to the wing, fuselage, landing gear, nacelle and propulsion system. Weight impact, of the engine change, on the remaining systems is assumed negligible.

Weight of the -5B propulsion system is 59,931 pounds (27,184 kg); 10,259 pounds (4,653 kg) less than the -5A baseline. The savings is due almost entirely to a reduction in engine and exhaust system weight. Differences in propulsion equipment are small. Net nacelle/inlet weight saving is 485 pounds (220 kg), comprising a 572 pound (260 kg) reduction in engine cowling and an 87 pound (39 kg) increase in the weight of the engine inlet. Savings in cowling weight results from a smaller engine envelope. The increase in inlet weight is due to increased capture area.

Differences in pylon and engine support weight are not included under nacelle/inlet weight. These differences, along with differences in the fuselage and wing, due to changes in load, are lumped under "Structural Weight Increment". This weight increment is supported by a structural/weight optimization analysis, details of which are discussed in the Structural Analysis Section.

Also included in Table 2-5 are weight reductions associated with a change in ground clearance, made possible by the smaller engine package. These savings comprise a 657 pound (298 kg) reduction in gear weight, resulting

TABLE 2-5
WEIGHT COMPARISON – CONFIGURATION –5B
(MINI-BYPASS) WITH –5A BASELINE (TURBOJET)
ENGLISH UNITS

CONFIGURATION	WEIGHT – POUNDS		
	5A TURBO-JET	5B MINI-BYPASS	DIFF
WING	75,347	75,245*	–102
H-TAIL	3,960	3,960*	0
V-TAIL	3,807	3,807*	0
FUSELAGE	47,713	47,689*	–24
LANDING GEAR	36,792	36,135	–657
FLIGHT CONTROLS	9,115	9,115	0
NACELLE/INLET	14,730	14,245	–485
PROPULSION (LESS FUEL SYSTEM)	70,190	59,931	–10,259
FUEL SYSTEM	3,820	3,820	0
EMERGENCY POWER UNIT	950	950	0
INSTRUMENTS	1,227	1,227	0
HYDRAULICS	5,684	5,684	0
PNEUMATICS	1,332	1,332	0
ELECTRICAL	4,850	4,850	0
NAVIGATION AND COMMUNICATIONS SYSTEM	2,756	2,756	0
FURNISHINGS	24,478	24,478	0
AIR CONDITIONING	4,854	4,854	0
ICE PROTECTION	489	489	0
HANDLING PROVISIONS	90	90	0
PENALTY-FLUTTER AND AEROELASTICITY	2,860	2,860	0
STRUCTURAL WEIGHT INCREMENT	–	–1,300	–1,300
MANUFACTURER'S WEIGHT EMPTY	315,044	302,217	–12,827
OPERATOR'S ITEMS	8,096	8,096	0
OPERATOR'S WEIGHT EMPTY	323,140	310,313	–12,827

*THE WEIGHT INCREMENT FOR STRENGTH ETC., FOR THESE ITEMS IS INCLUDED UNDER THE ITEM STRUCTURAL WEIGHT INCREMENT AND LISTED SEPARATELY.

TABLE 2-5
WEIGHT COMPARISON – CONFIGURATION –5B
(MINI-BYPASS) WITH –5A BASELINE (TURBOJET)
METRIC UNITS

CONFIGURATION	WEIGHT – KILOGRAMS		
	5A TURBO-JET	5B MINI-BYPASS	DIFF
WING	34,177	34,131	–46
H-TAIL	1,796	1,796	0
V-TAIL	1,727	1,727	0
FUSELAGE	21,642	21,631	–11
LANDING GEAR	16,689	16,391	–298
FLIGHT CONTROLS	4,134	4,134	0
NACELLE/INLET	6,681	6,461	–220
PROPULSION (LESS FUEL SYSTEM)	31,838	27,185	–4653
FUEL SYSTEM	1,733	1,733	0
EMERGENCY POWER UNIT	431	431	0
INSTRUMENTS	557	557	0
HYDRAULICS	2,578	2,578	0
PNEUMATICS	604	604	0
ELECTRICAL	2,200	2,200	0
NAVIGATION AND COMMUNICATIONS SYSTEM	1,250	1,250	0
FURNISHINGS	11,103	11,103	0
AIR CONDITIONING	2,202	2,202	0
ICE PROTECTION	222	222	0
HANDLING PROVISIONS	41	41	0
PENALTY-FLUTTER AND AEROELASTICITY	1,297	1,297	0
STRUCTURAL WEIGHT INCREMENT	–	–	–590
MANUFACTURER'S EMPTY WEIGHT	142,902	137,084	–5818
OPERATIONAL ITEMS	3,672	3,672	0
OPERATIONAL EMPTY WEIGHT	146,574	140,756	–5818

from a 14 inch (35.5 cm) reduction in strut length, a 102 pound (46 kg) reduction in wing weight, and a 24 pound (11 kg) reduction in fuselage weight because of smaller gear door areas.

The mini-bypass engine/nacelle installation is 10,744 pounds (4873 kg) [propulsion 10,259 lb. (4,653 kg) and nacelle/inlet 485 lb. (220 kg)] lighter than the turbojet baseline, and has a nacelle c.g. 16 inches (41 cm) further forward compared to the baseline turbojet. This c.g. change results primarily from installation of the shorter engine, since the mean inlet location of both engines is at about the same location (Sta. 2500). Combining the lighter engine installation, moved 16 inches (41 cm) forward, with the structural weight reduction, due to reduced loads and ground clearance, moves the weight empty c.g. of the airplane 19.6 inches (50 cm) forward.

The total saving in airplane Operating Weight Empty from the -5A configuration is 12,827 pounds (5,818 kg).

Aerodynamics Analysis

The trimmed lift and drag characteristics for the P7 powered -5B design are obtained by adjusting the wave drag of the baseline turbojet powered -5A design for the difference due to the P7 nacelles. The difference in nacelle skin friction drag is accounted for in the installed propulsion system performance. Results of the wave drag program indicate a reduction in supersonic wave drag of 3.0 counts (ΔC_D = .00030) due to the differences in nacelle shape and location. The characteristics used to determine the mission performance for the P7 powered aircraft are obtained by subtracting this increment from the wave drag of the baseline turbojet powered design.

Performance Results

Estimated airplane performance characteristics for the P7 powered design are presented in Figures 2-33 through 2-35 as a function of engine size. The mission profile and reserve ground rules are the same as used for the baseline -5A aircraft (Figure 1-20). The takeoff gross weight is held constant at 750,000 lb. (340,194 kg) and the payload is fixed at 55,965 lb. (25,385 kg).

Figure 2-33 presents the takeoff characteristics and the height above the runway at 3.5 n.mi. (6.5 km) from the start of takeoff, with the throttle cut back to meet the 4 percent all-engine-climb-gradient requirement of FAR Part 36. The characteristics of the aircraft with the engine size selected as described in the Engine Sizing Section are indicated on the figure and, for reference, characteristics of the -5A baseline are also shown.

Figure 2-34 presents the variation of operator's weight empty with engine size used for the mission performance calculations, the altitude for maximum range factor at the start of the 2.2 M cruise, and the mission range.

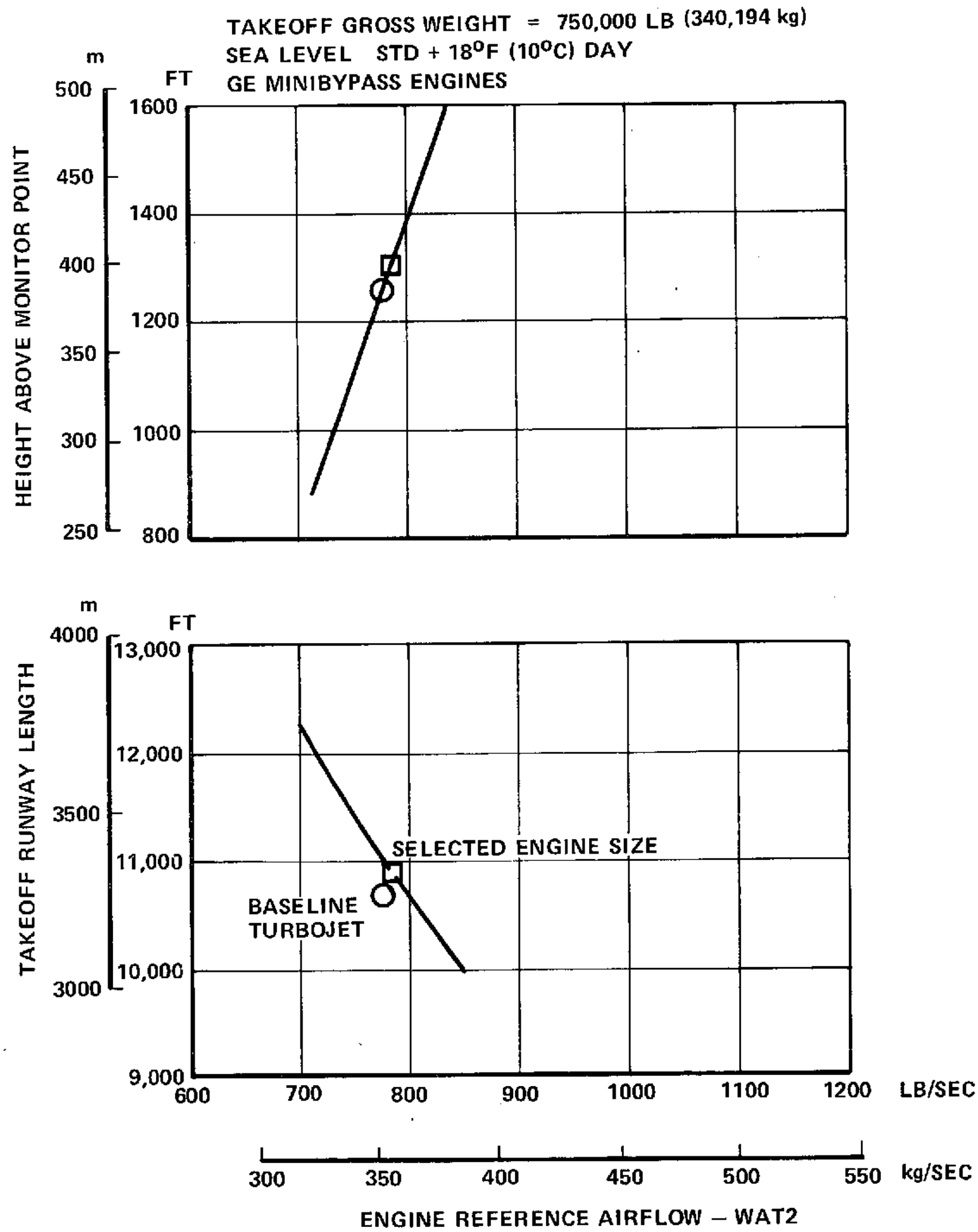


FIGURE 2-33. EFFECT OF ENGINE SIZE ON TAKEOFF PERFORMANCE

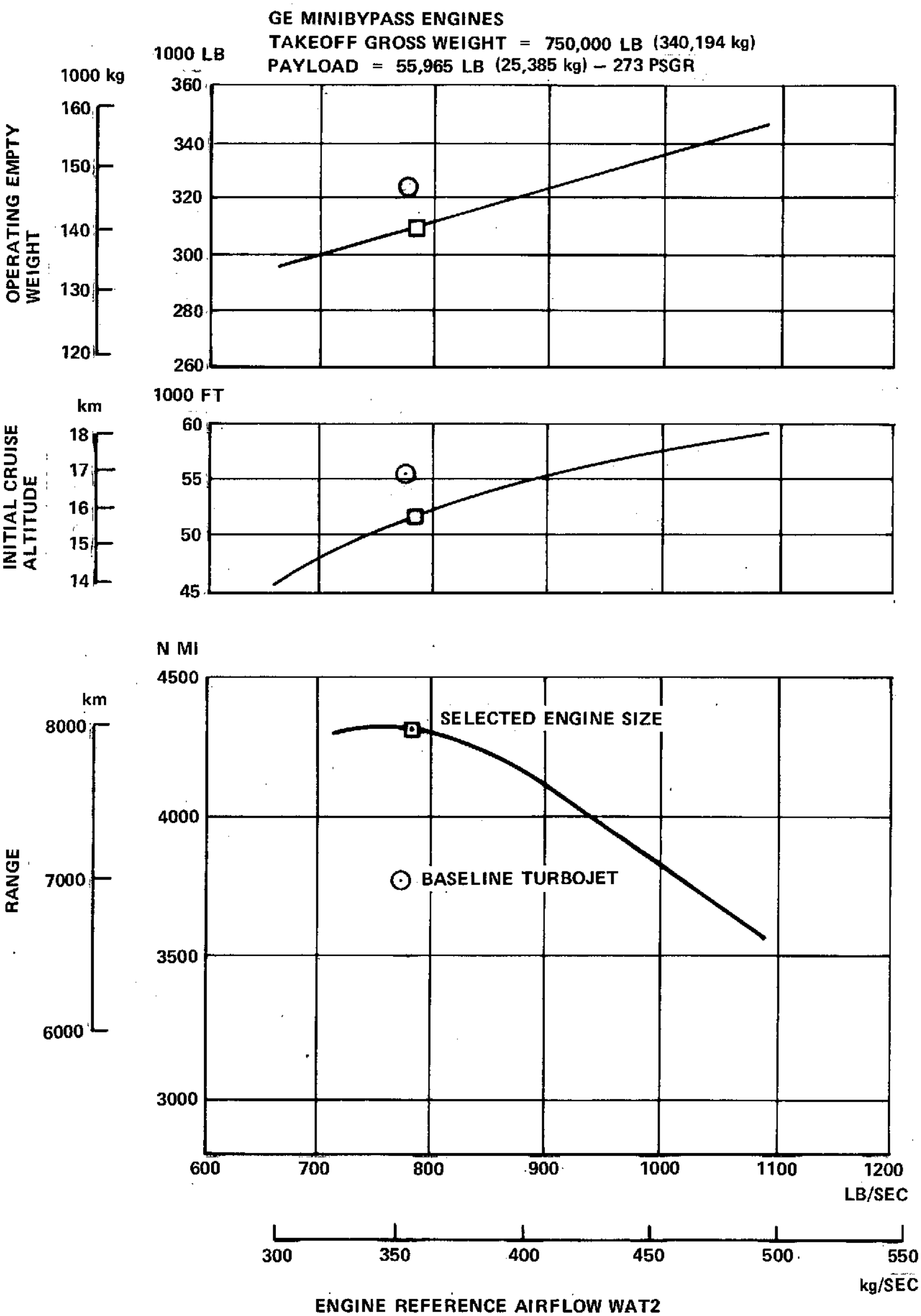


FIGURE 2-34. EFFECT OF ENGINE SIZE ON MISSION PERFORMANCE

The selected engine size as indicated in the figure is the same as initially selected in the engine sizing paragraph. Note that this size engine provides the best range.

Figure 2-35 presents some of the details of the effect of engine size on the optimum cruise L/D, cruise installed SFC, and the 2.2 M cruise range factor.

The data presented in the last two figures accounts for the changes with engine size of engine and nacelle weight, and inlet and nacelle drags, but neglects the changes in aircraft wave drag. For a ten percent change in engine size, this effect is quite small, but, for example, with the P7 engine airflow increased from 782 lb/sec (355 kg/sec) to 1013 lb/sec (459 kg/sec) the wave drag increases by 1.3 counts (ΔC_D = .00013), which further reduces the range with that size engine by about 50 n.mi. (93 km).

The performance for the P7 powered design with the 782 lb/sec (355 kg/sec) engine is summarized below:

Takeoff Gross Weight	750,000 lb (340,194 kg)
Payload	55,965 lb (25,385 kg)
Takeoff Field Length	10,850 ft (3,307 m)
Height at 3.5 n.mi. (6.5 km) Takeoff Point	1,292 ft (394 m)
Range	4,308 n.mi. (7979 km)
Initial Cruise Altitude	52,213 ft (15.9 km)
Direct Operating Cost (1973 $)	1.74 cents/seat n.mi.

The effect of initial subsonic leg on the mission range is shown in Figure 2-36. With a 600 n.mi. (1112 km) subsonic leg, the range is reduced by 5 percent.

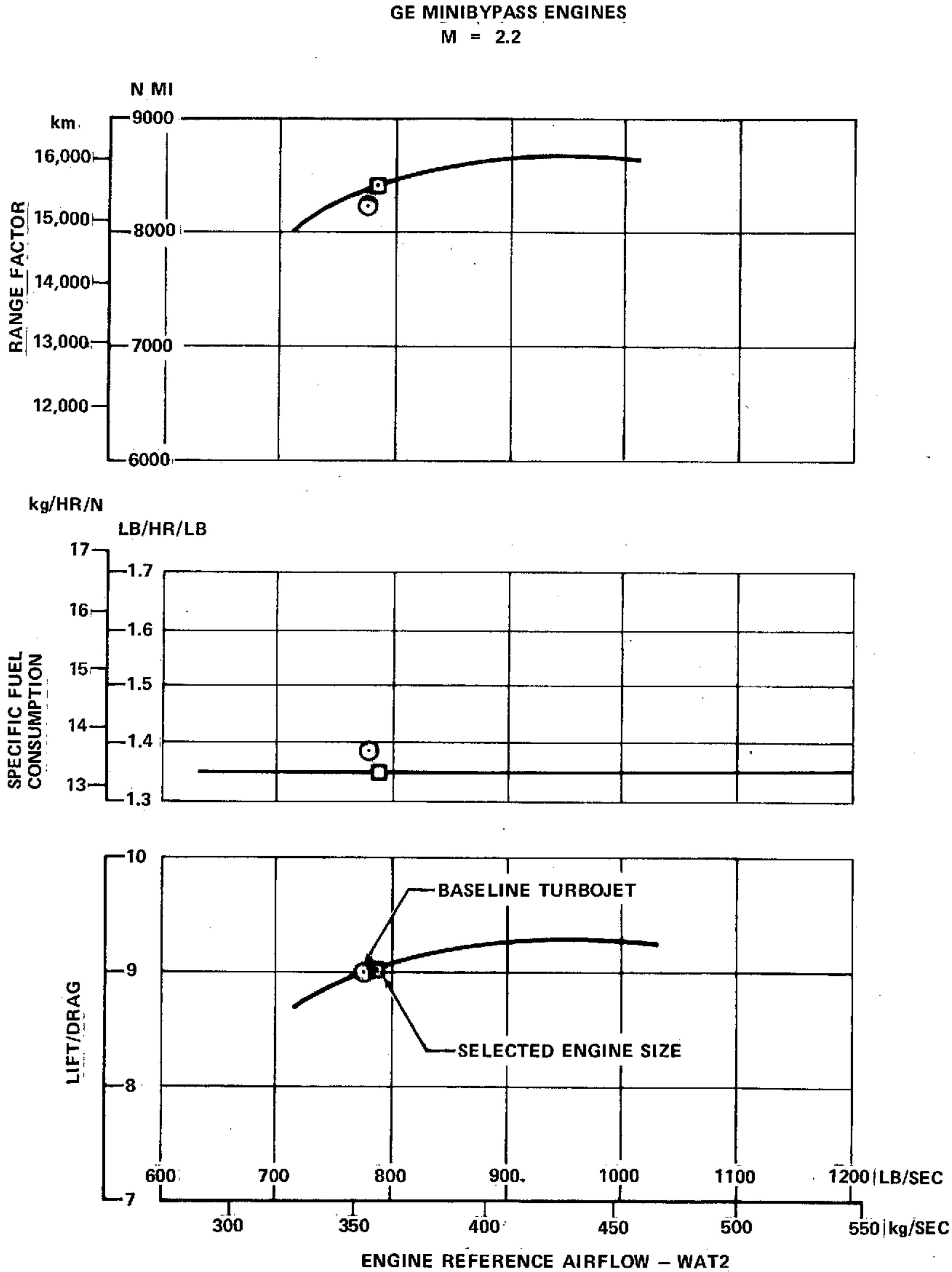


FIGURE 2-35. EFFECT OF ENGINE SIZE ON CRUISE PARAMETERS

Engine/Suppression Analysis

A preliminary study has been completed to assess the effect on performance of using the DAC exhaust system (nozzle/reverser/suppressor) in place of the GE exhaust system. Sizing criteria for the engine is takeoff thrust [52,000 lb. (231.3 kN) per engine, suppressed, uninstalled)], suppressor temperature limit of 1500°F (1089°K) and FAR Part 36 noise [sea level, 0.3 Mach, 2270 ft. (747 m) sideline and 1050 ft (320 m), 0.3 Mach, takeoff/cutback, Std. + 18°F (10°C) day]. The acoustics data for the DAC suppressor is estimated to be 12 PNdB in flight suppression at an exhaust velocity of 2570 ft/sec. (782 m/sec). For this condition the assumption is made that the net thrust loss is 8 percent. These assumptions seem realistic and future testing is expected to confirm these levels. Utilizing these assumptions Figure 2-37 is generated. Different suppressor capabilities and different suppressor design point velocities have been used. With a 12 PNdB suppressor designed at 2500 ft/sec (760 m/sec) exhaust velocity the engine size is determined to be 773 lb/sec (350.6 kg/sec) inlet corrected airflow.

Dimensions for this engine and nacelle are shown in Figures 2-38 and 2-39. Scaling factors as supplied by GE are used as follows:

Dimension Scaling Factors

$$D_2 = D_1 \left(\frac{WAT2}{773}\right)^{0.5}$$

$$L_2 = L_1 \left(\frac{WAT2}{773}\right)^{0.5}$$

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SUPPRESSOR PERFORMANCE AT TAKEOFF FOR FAR PART 36
(TRADED, SIDELINE, AND CUTBACK)

SEA LEVEL, 0.3M, STD + 18° + (10°C) DAY
F_N/ENGINE = 52,000 LB (231.3 kN) SUPPRESSED, UNINSTALLED

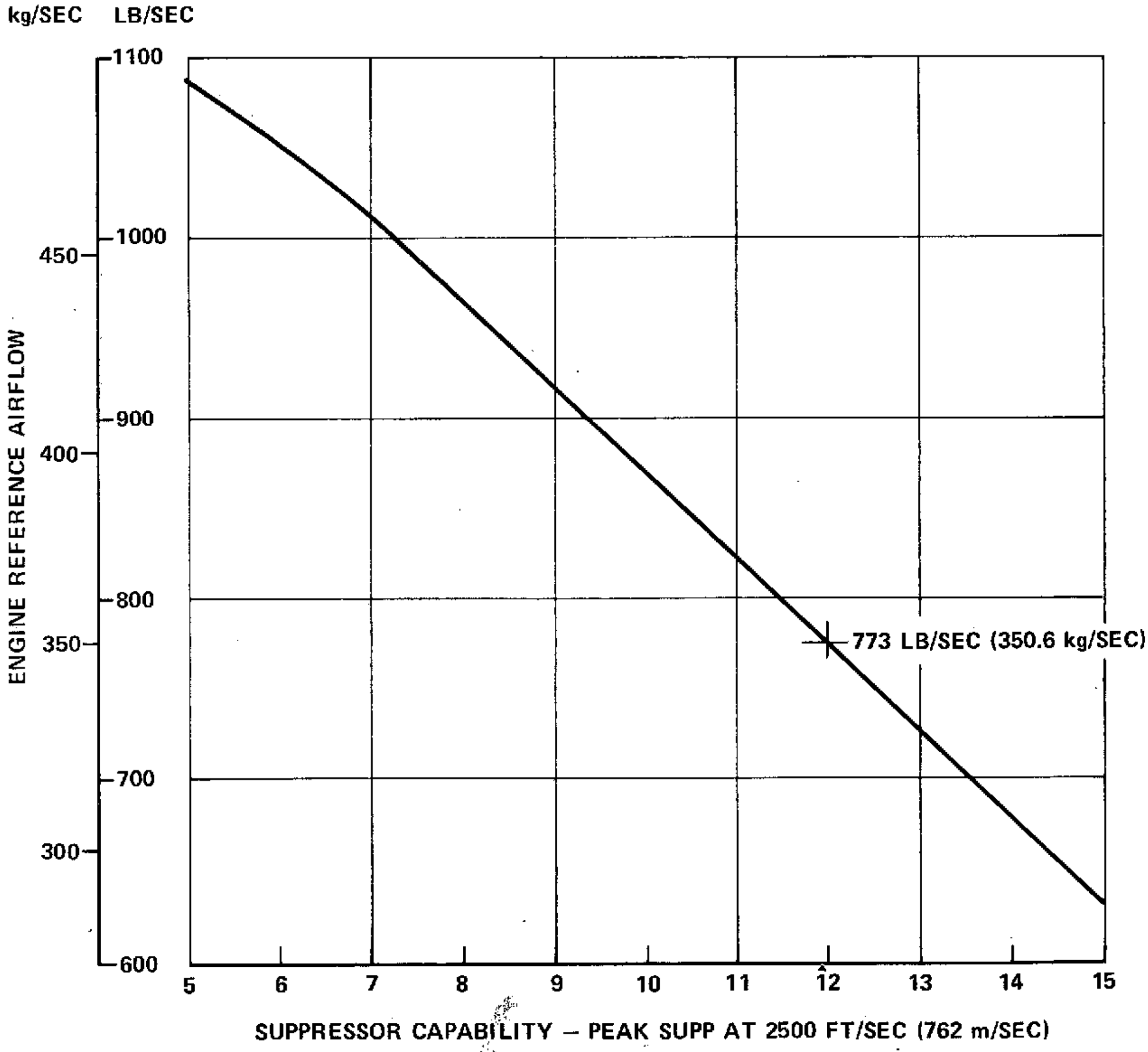
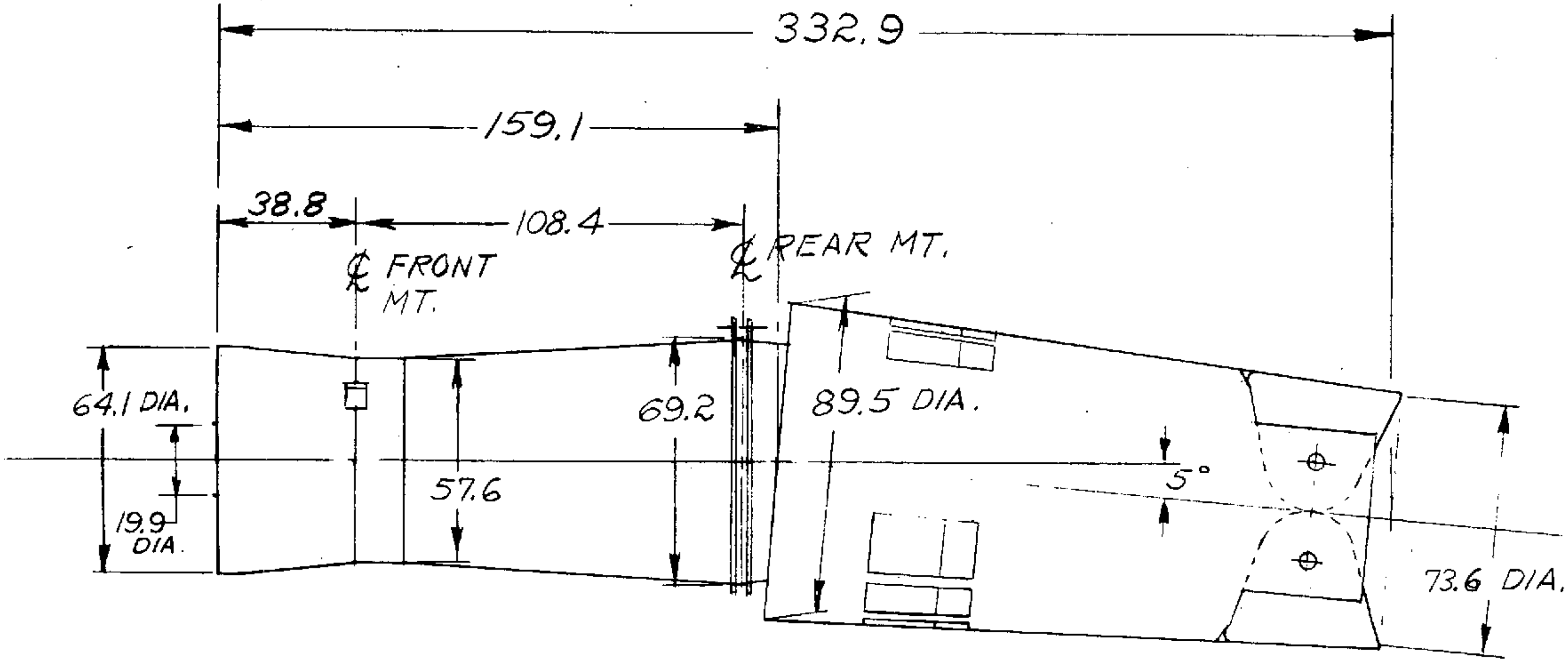


FIGURE 2-37. GE MINI-BYPASS TURBOJET WITH DAC NOZZLE/SUPPRESSOR



AIRFLOW 773 LBS/SEC (350.6 kg/SEC)

FIGURE 2.38. GE MINI-BYPASS-ENGINE WITH DAC NOZZLE

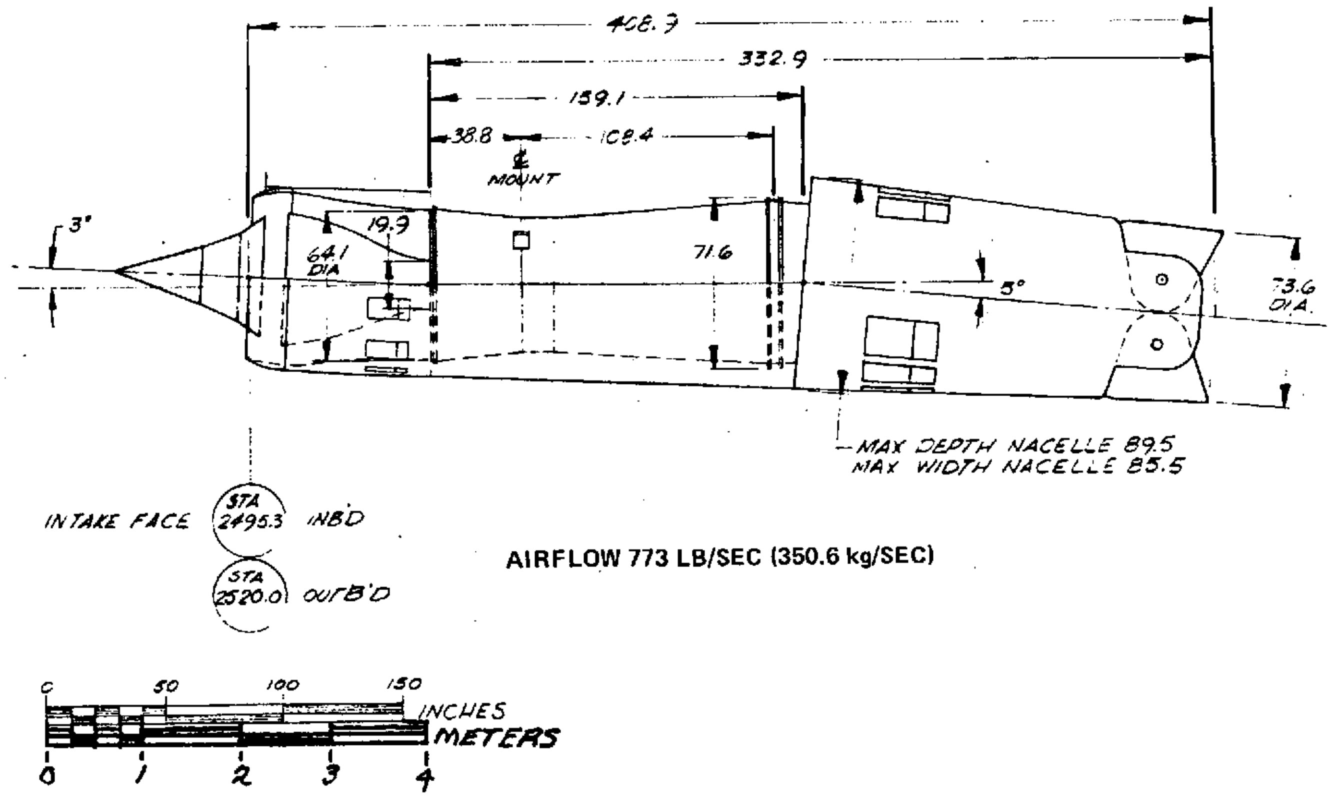


FIGURE 2.39. GE MINI-BYPASS ENGINE/DAC NOZZLE INSTALLATION

Weight

Base Engine Weight	11311 lb. (5131 kg)
Nozzle + Reverser	2432 lb. (1103 kg)
Noise Suppressor	1607 lb. (729 kg)
TOTAL	15350 lb. (6963 kg)

Scaling Factor

Engine: $W_2 = W_1 \left(\frac{WAT2}{773}\right)^{1.2}$

Nozzle, Reverser, Suppressor: $W_2 = W_1 \left(\frac{WAT2}{773}\right)^{1.16}$

Airplane Performance Results

The operating weight of the P7 configuration (-5B), covered in Section 2, would have to be increased by 5000 lb. (2268 kg) if the DAC exhaust system were used in lieu of the GE system. The nacelle must be enlarged to accommodate the ejector and make provisions for stowing the suppressor. After accounting for the incremental effects of wave drag, nacelle skin friction drag, nozzle velocity coefficient, and weight the estimated range is approximately 4000 n.miles (7410 km).

Conclusion

Considering FAR Part 36 or Part 36 minus two noise levels as AST requirements, the mini-bypass engine remains a strong contender as a candidate engine for the airplane. Completion of development testing is required prior to selection of the specific suppressor type. The DAC type ejector/suppressor appears to be the more efficient; however, the GE plug nozzle/multi-chute type even with some degradation can be tolerated with the mini-bypass engine.

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PRELIMINARY ENGINE SCREENING STUDY

Cycle Analysis

During the 1973 NASA AST technology studies the duct heating turbofan engine, as offered by Pratt & Whitney Aircraft (P&WA), was shown to be one of the preferred engine cycles for a Mach 2.2 cruise aircraft. In-house cycle studies established the desired cycle parameters and the amount of augmentation required for a duct heating turbofan engine best suited for a Mach 2.2 cruise design. Results of these studies are used to guide the duct heating turbofan selection. The selected duct heating turbofan is used for evaluation in the baseline airplane design.

The matrix of engine cycles selected for this screening study is identified as follows:

Sea Level, 0.3 Mach Design Point

° Bypass Ratio	1.5, 2.0, 2.5
° Cycle Pressure Ratio	12, 15, 18, 21, 24
° Turbine Inlet Temperature, °F (°K)	2600 (1700), 2800 (1811) (at takeoff)
° Fan Pressure Ratio	3.3 (3 stage fan)
° Duct Temperature, °F (°K)	500 (533), 1000 (811), 1500 (1089)

The engines are sized and configured to meet takeoff thrust requirements and FAR Part 36 sideline noise levels and verified for adequate supersonic cruise requirements. The installed takeoff thrust required is 48,000 lb/engine (213.5 kN/eng) [sea level, 0.3 Mach, Std. + 18°F (10°C) day], based on the 1973 DAC AST technology study baseline airplane requirements. For suppression, a 5 EPNdB jet noise suppressor is assumed on the fan stream only, with no thrust penalty as defined in the 1973 P&WA study. The primary stream is restricted in velocity so as not to exceed 97 EPNdB sideline jet noise [sea level, 0.3 Mach, 2100 ft. (640 m) sideline, Std. + 18°F (10°C) day], thereby alleviating the requirement for a jet noise suppressor on the primary stream. Figure 3-1 shows primary jet velocity as a function of bypass ratio and overall cycle pressure ratio for two design turbine inlet temperatures at takeoff. For constant 97 EPNdB primary stream jet noise, bypass ratio is shown to vary from 1.6 to 2.13 as a function of cycle pressure ratio and turbine inlet temperature. Figure 3-2 illustrates the engine sizing logic for a representative cycle based on engine size, P&WA sideline noise from their Phase I Task I AST studies, 5 EPNdB fan stream suppressor and the takeoff thrust requirement of 48,000 lb/engine (213.5 kN/eng). For FAR Part 36 sideline noise the engine size is 920 lb/sec (417.3 kg/sec) design corrected airflow with the fan stream augmented to 1040°F (833°K).

Cruise performance for the study matrix is examined for Mach 2.2 at 52,000 ft. (15,850 m). The airflow schedule used to correlate cruise with takeoff performance, shown in Figure 3-3a, closely matches that used by the engine companies in their AST engine studies. Fan pressure ratio is fixed at 2.4 for cruise as compared to 3.3 for takeoff, again consistent with engine company AST data.

Cycle pressure ratio selection for cruise is influenced by consideration of a nominal compressor discharge temperature limit of 1100°F (867°K). This temperature reflects consideration of material selection and life for the compressor final stages and the cooling airflow requirements to the turbine. Figure 3-3b shows that at cruise, a compression ratio of 10 correlates with a compressor discharge temperature of 1100°F (867°K). Compression ratios of 8 and 12 are also examined to determine sensitivity of cruise thrust and SFC to compression ratio.

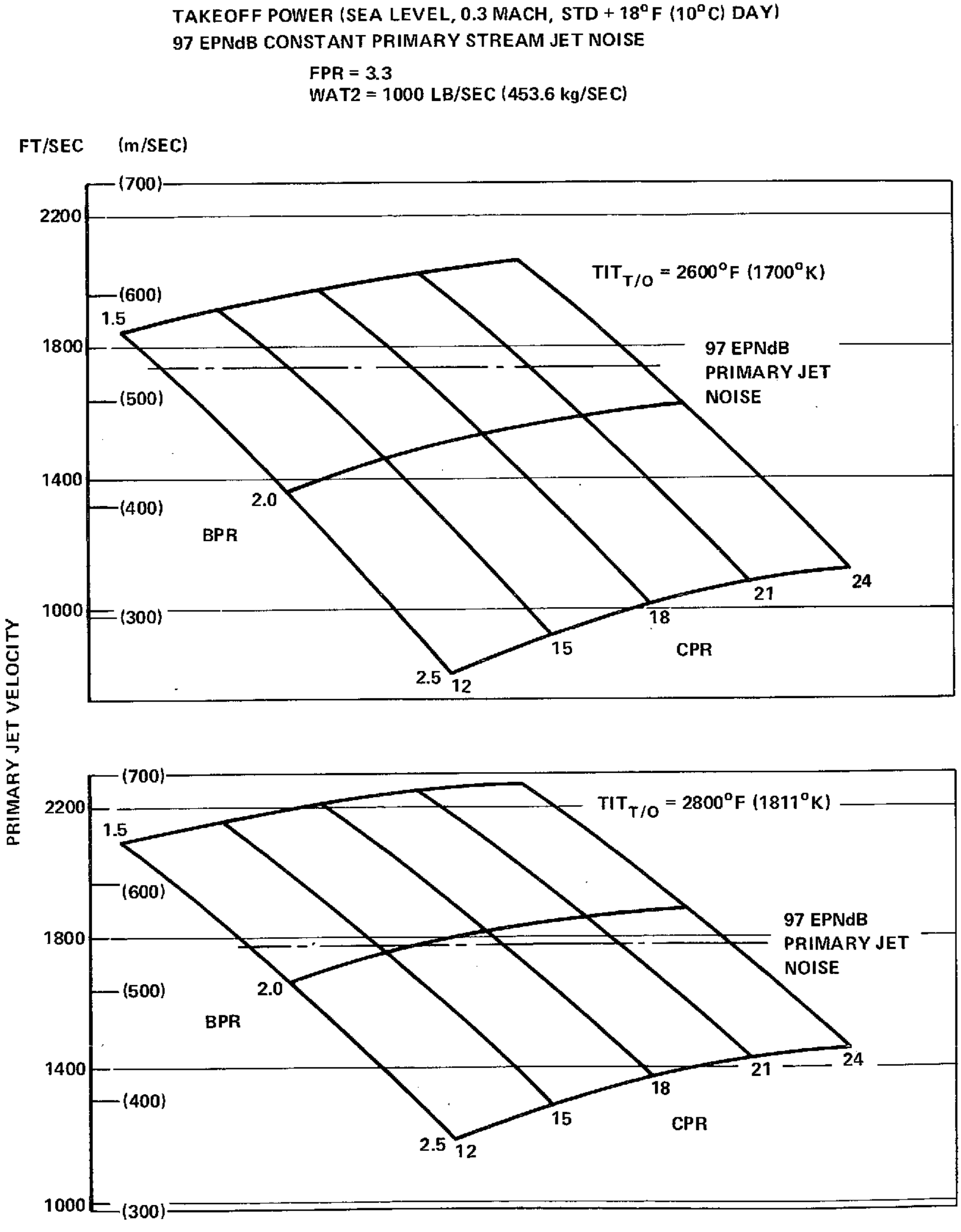


FIGURE 3-1. DUCT HEATING TURBOFAN CYCLE SELECTION

DUCT HEATING TURBOFAN

SEA LEVEL, 0.3 M, 2100 FT (640.1m) SIDELINE, STD + 18°F (10°C) DAY
F_N REQUIRED = 48,000 LB/ENGINE (213.51 kN) (UNINSTALLED, SUPPRESSED)

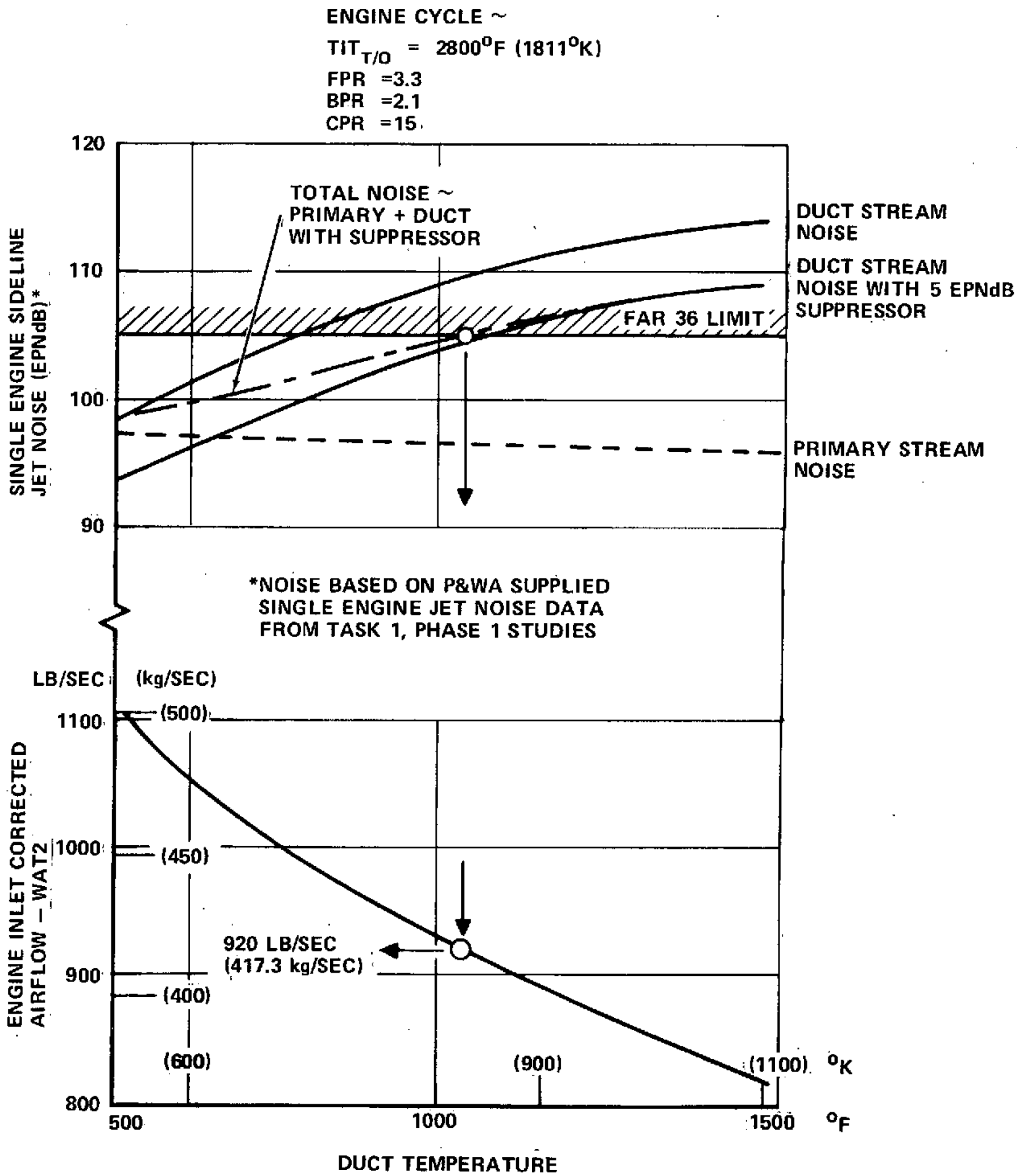
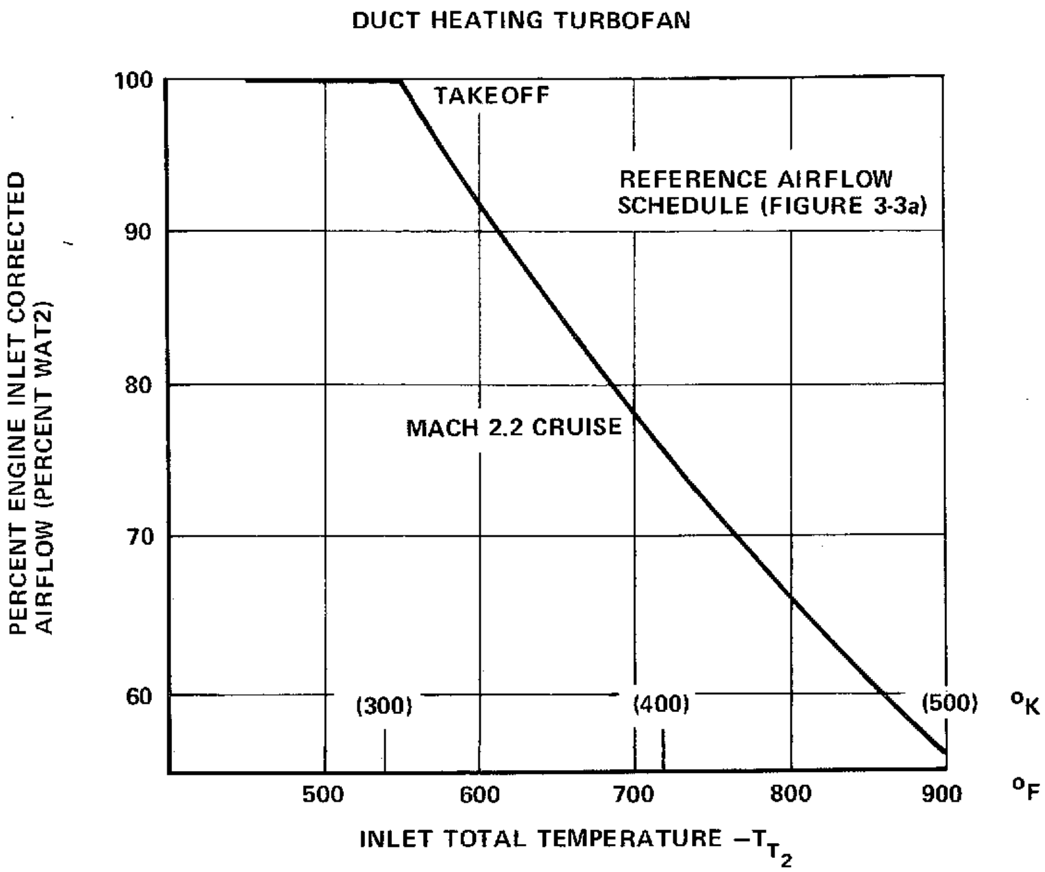


FIGURE 3-2. ENGINE SIZING AT TAKEOFF FOR FAR PART 36



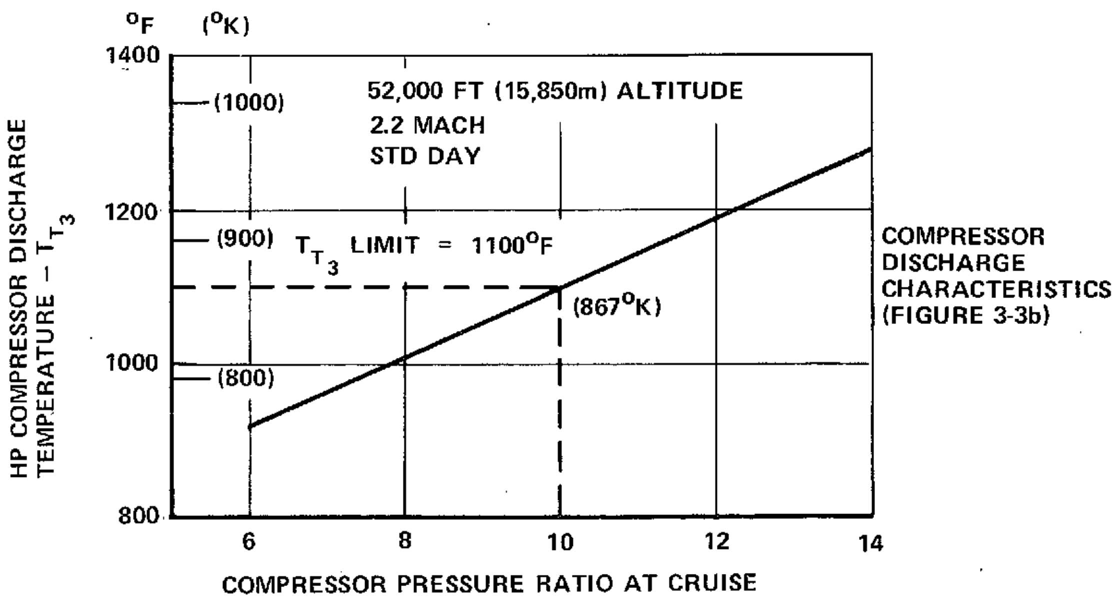


FIGURE 3-3. CYCLE OPERATING CONSTRAINTS

Turbine inlet temperature at max cruise power is established at 200°F (111°K) less than at max takeoff, consistent with AST engine rating philosophy used at the engine companies. Fan stream augmentation up to 1500°F (1089°K) is considered. Bypass ratio at cruise is determined to be nearly the same as at takeoff. An appropriate correction is made for nacelle drag.

Figure 3-4a shows installed SFC versus thrust at Mach 2.2 cruise for variations in design compression ratio of 12 to 24. Compression ratio for cruise is maintained at 10 and turbine inlet temperature at 2400°F (1589°K). Examination of cruise SFC at the desired engine thrust level of 16,000 to 18,000 lbs. (711.7 to 889.6 kN) shows that design compression ratio has small impact on minimum cruise SFC, ranging from 1.58 to 1.61 lb/hr/lb (.161 to .164 kg/hr/N).

Figure 3-4b shows sensitivity of cruise performance for variations in cruise compression ratio from 8 to 12. The effect on SFC is small, varying from 1.58 to 1.63 lb/hr/lb (.161 to .166 kg/hr/N).

Figure 3-4c shows that SFC remains virtually unchanged as max cruise turbine inlet temperature increases from 2400 to 2600°F (1589 to 1700°K).

Conclusions

The results show that duct augmentation at cruise overrides variations in cycle parameters, thereby diminishing the criticality of cycle parameter selection at cruise, leaving noise and mission requirements during takeoff dominant for cycle selection.

The results show modest duct augmentation, below 1500°F (1089°K), required for cruise. Even less augmentation is required for takeoff, 1040°F (833°K) for FAR Part 36 sideline noise.

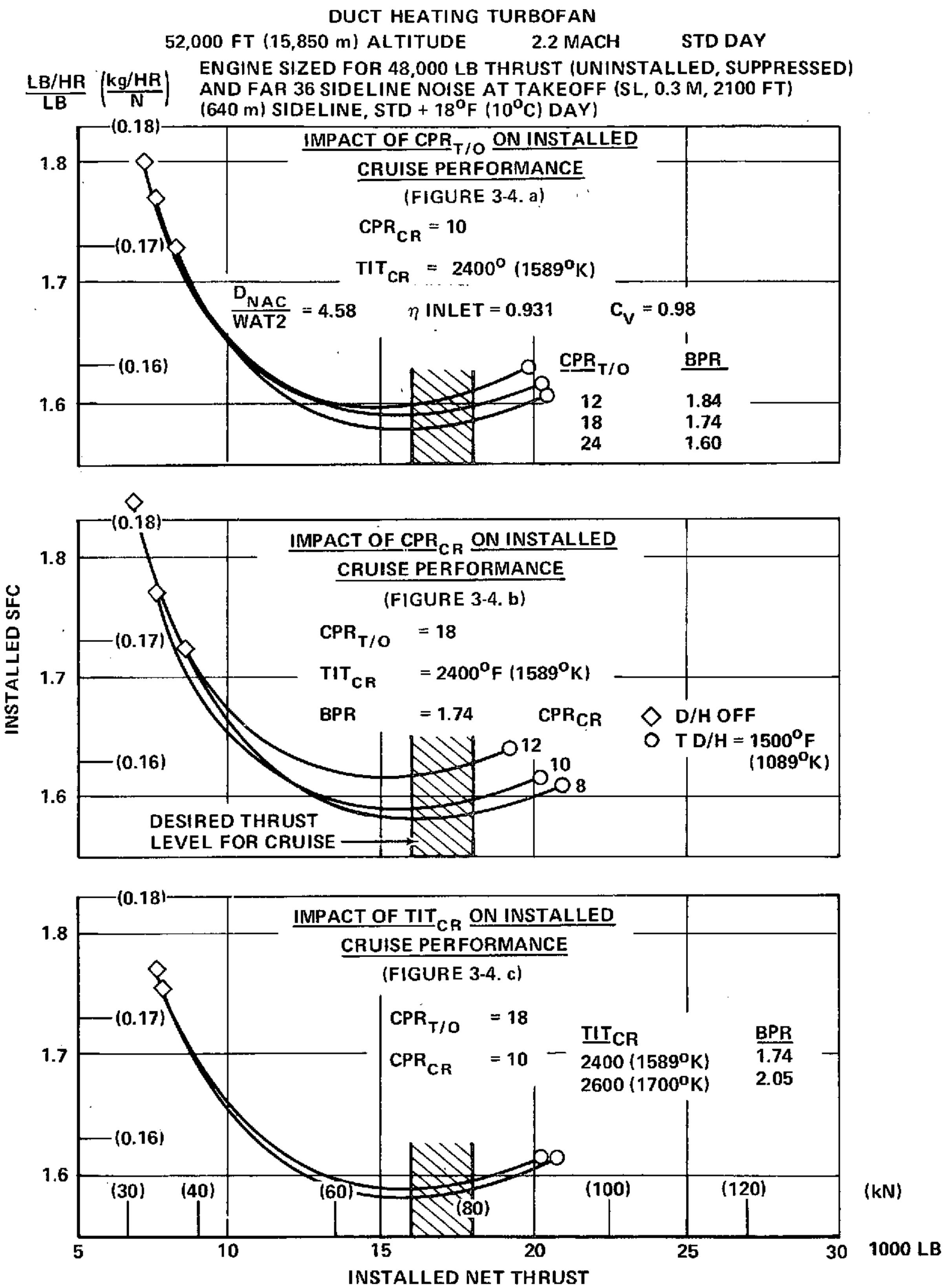


FIGURE 3-4. IMPACT OF CYCLE PARAMETER VARIATION ON INSTALLED CRUISE PERFORMANCE

Results of this study were used to guide selection of a duct heating turbofan for the airplane integration study. The engine selected, with NASA concurrence, was the P&WA Phase II Mach 2.4 duct heating turbofan 501, reconfigured to DAC Mach 2.2 installation requirements and identified as the 501D. A data package on the 501D was supplied by P&WA through their NASA-Lewis engine study contract.

General Analysis

Engine data for sizing and performance are based on the P&WA duct heating turbofan engine cycle identified as the -501D. The previous section identified a cycle of this type as a preferred duct heating turbofan cycle selected for further integration studies. This engine shown in Figure 3-5 is referred to by P&WA as a variable stream control engine.

Sizing criteria for this engine is takeoff thrust [52,000 lb. (231.3 kN) per engine, uninstalled, suppressed, no external drag], suppressor temperature limit [1200°F (922°K) for chute type and 1500°F (1089°K) for finger type, per P&WA] and FAR Part 36 noise [sea level, 0.3 Mach, 2270 ft. (692 m) sideline and 1050 ft. (320 m), 0.3 Mach, takeoff/cutback, Std. + 18°F (10°C) day].

Figure 3-6 illustrates the engine sizing logic based on P&WA suppressor temperature limits, suppressor type and characteristics, engine airflow and velocity, and four engine unsuppressed sideline noise (DAC calculations). Data are shown for no suppressor, a chute type suppressor and a finger type suppressor for various duct heat temperatures. P&WA suppressor loss data is used to determine takeoff thrust required (see Figure 3-7). Schematics of the chute and finger suppressor configurations are shown in Figure 3-8. As shown in Figure 3-6, the minimum size solution is an 875 lb/sec (397 kg/sec) inlet corrected airflow engine providing 54,500 lb. (242.4 kN) of thrust [52,000 lb. (231.3 kN) suppressed] at S.L., 0.3 M, standard + 18°F (10°C) day with a 5.1 PNdB finger type suppressor on the fan stream.

At the takeoff/cutback point, 33,250 lb. (147.9 kN) of thrust, the fan stream velocity is too low to gain benefit from the suppressor, see Figure 3-9. Further, as shown in the figure, the unsuppressed jet noise is 107.4 to 108.9 EPNdB depending on the aircraft altitude over the 3.5 n.mi. (6.5 km) noise monitor.

P&WA DUCT HEATING TURBOFAN
(VARIABLE STREAM CONTROL ENGINE)
DH/TF-501 CONCEPT

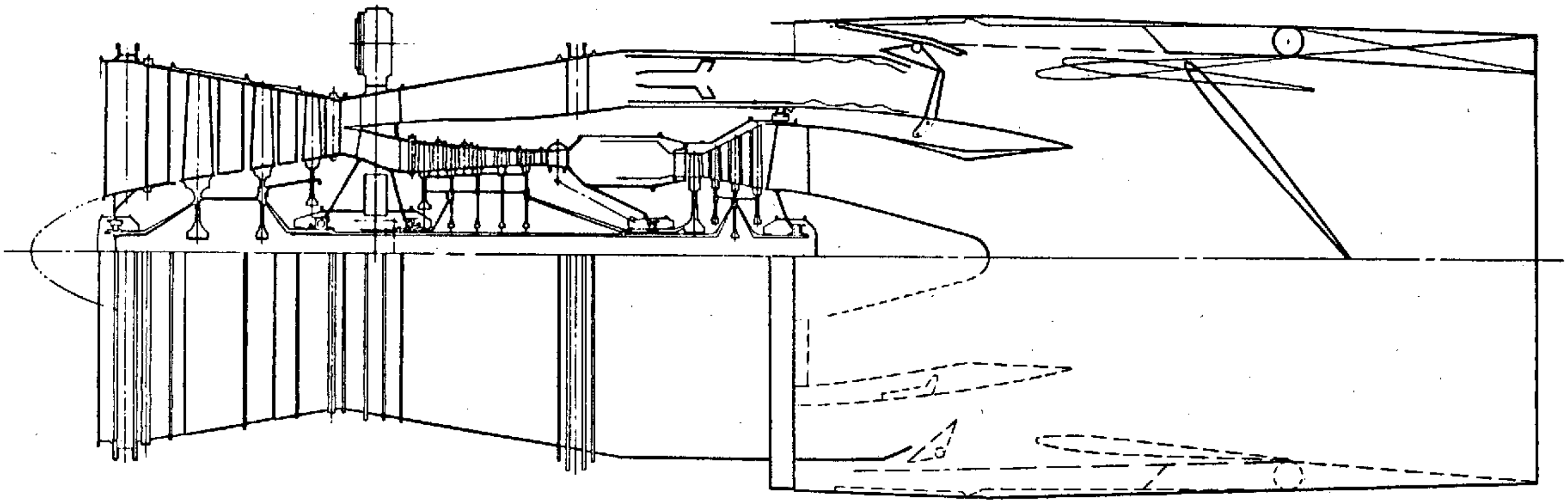


FIGURE 3-5. ENGINE SCHEMATIC

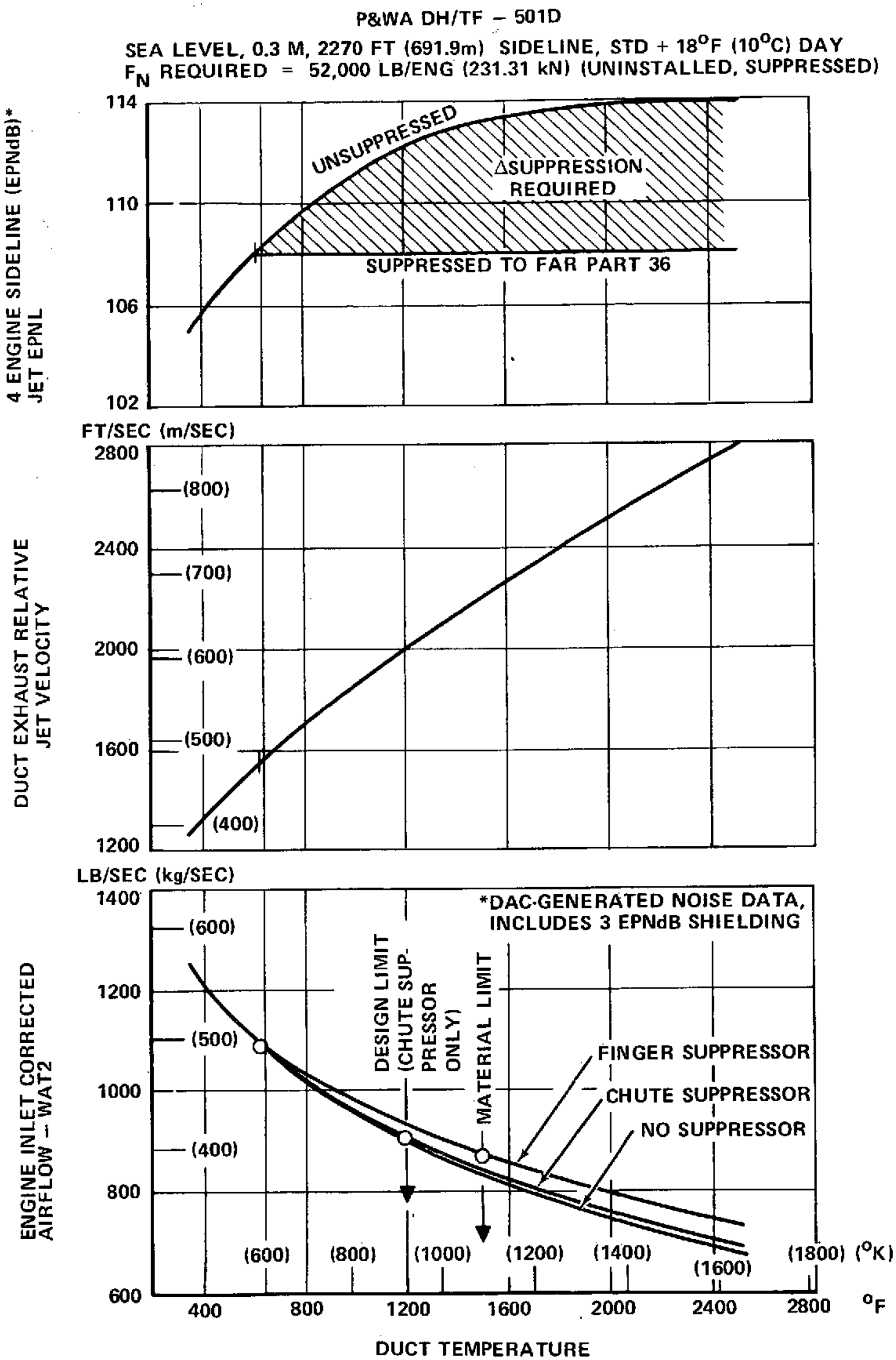


FIGURE 3-6. ENGINE SIZING FOR TAKEOFF

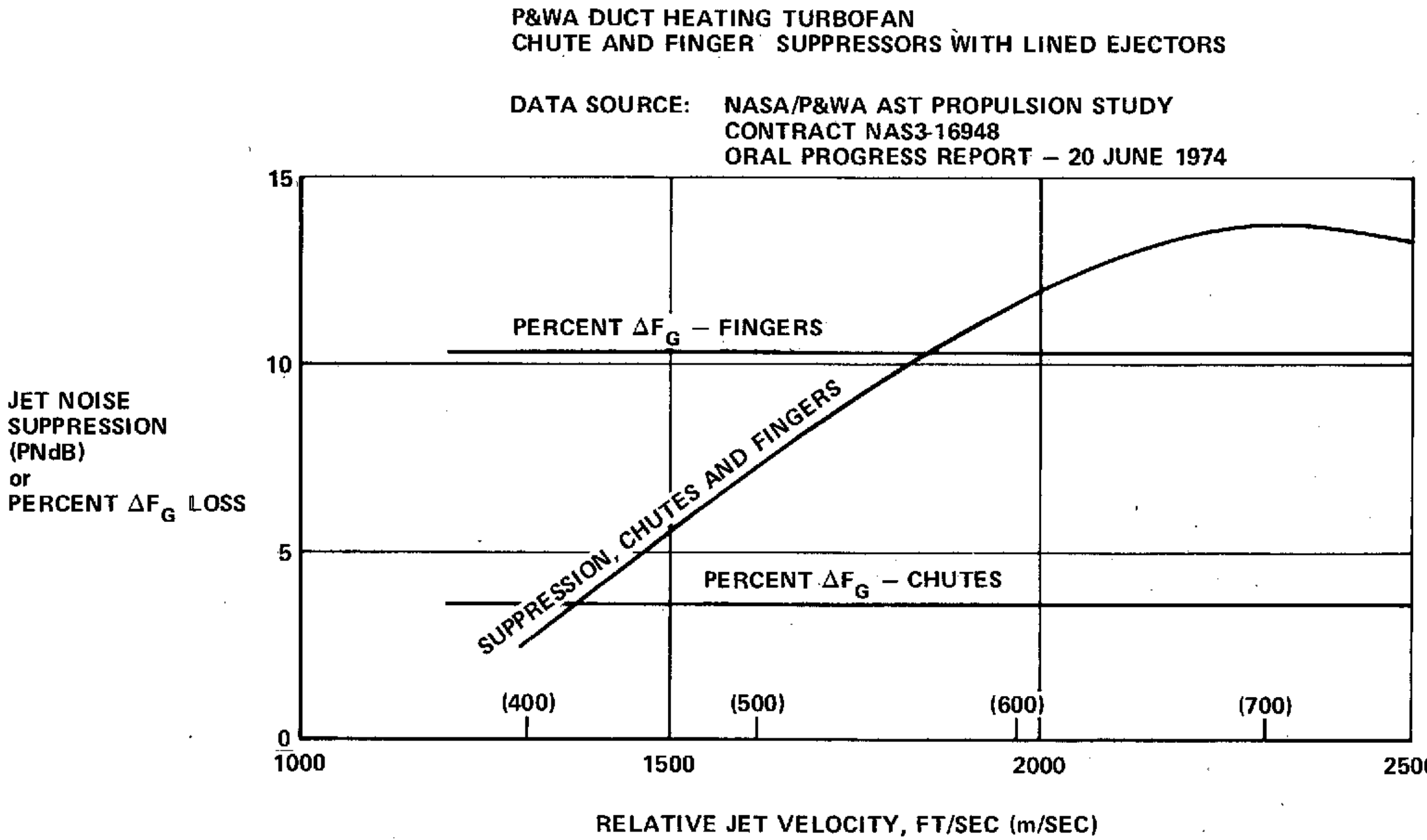
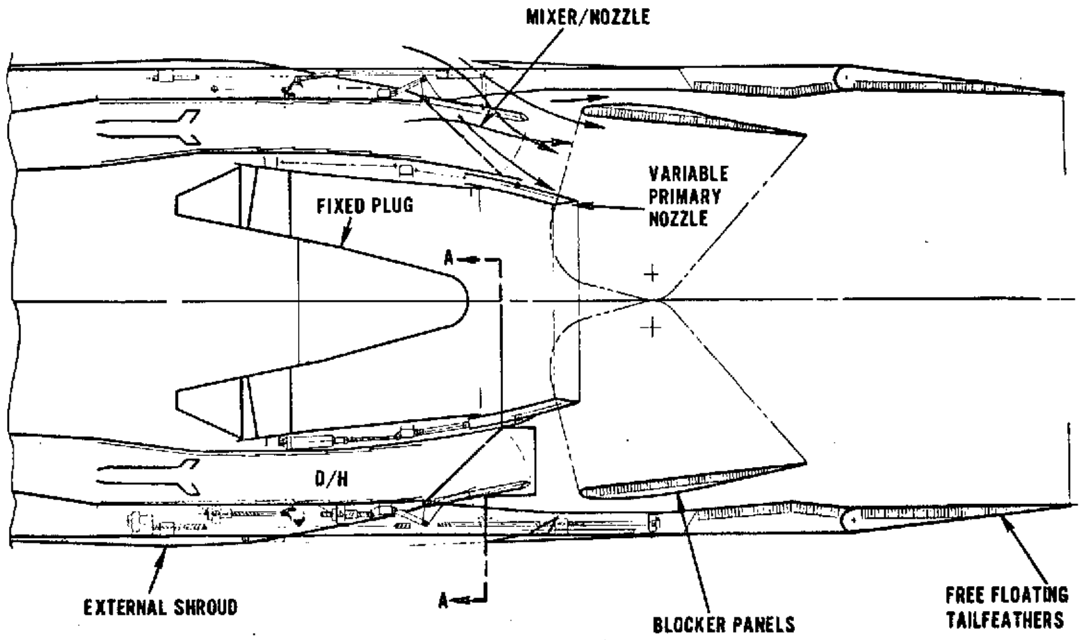


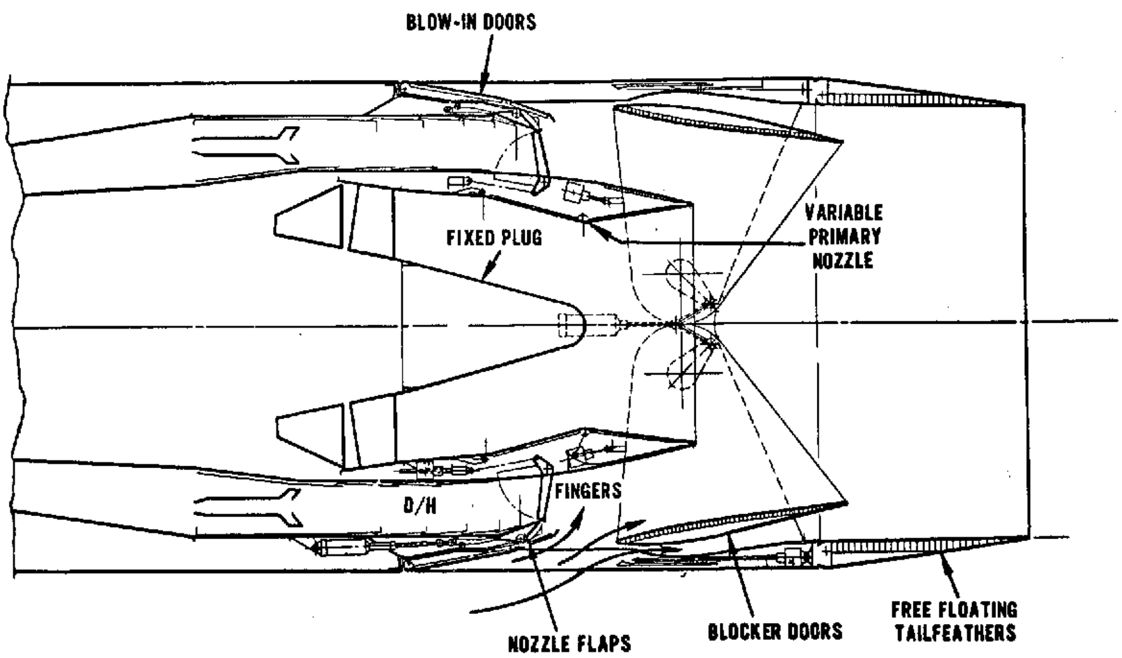
FIGURE 3-7. JET NOISE SUPPRESSOR CHARACTERISTICS

DUCT HEATING TURBOFAN

Data Source: NASA/P&WA AST Propulsion Study
Contract NAS3-16948
Oral Progress Report - 20 June 1974



CHUTE TYPE SUPPRESSOR



FINGER TYPE SUPPRESSOR

FIGURE 3-8. P&WA EXHAUST SYSTEM/SUPPRESSOR CONCEPTS

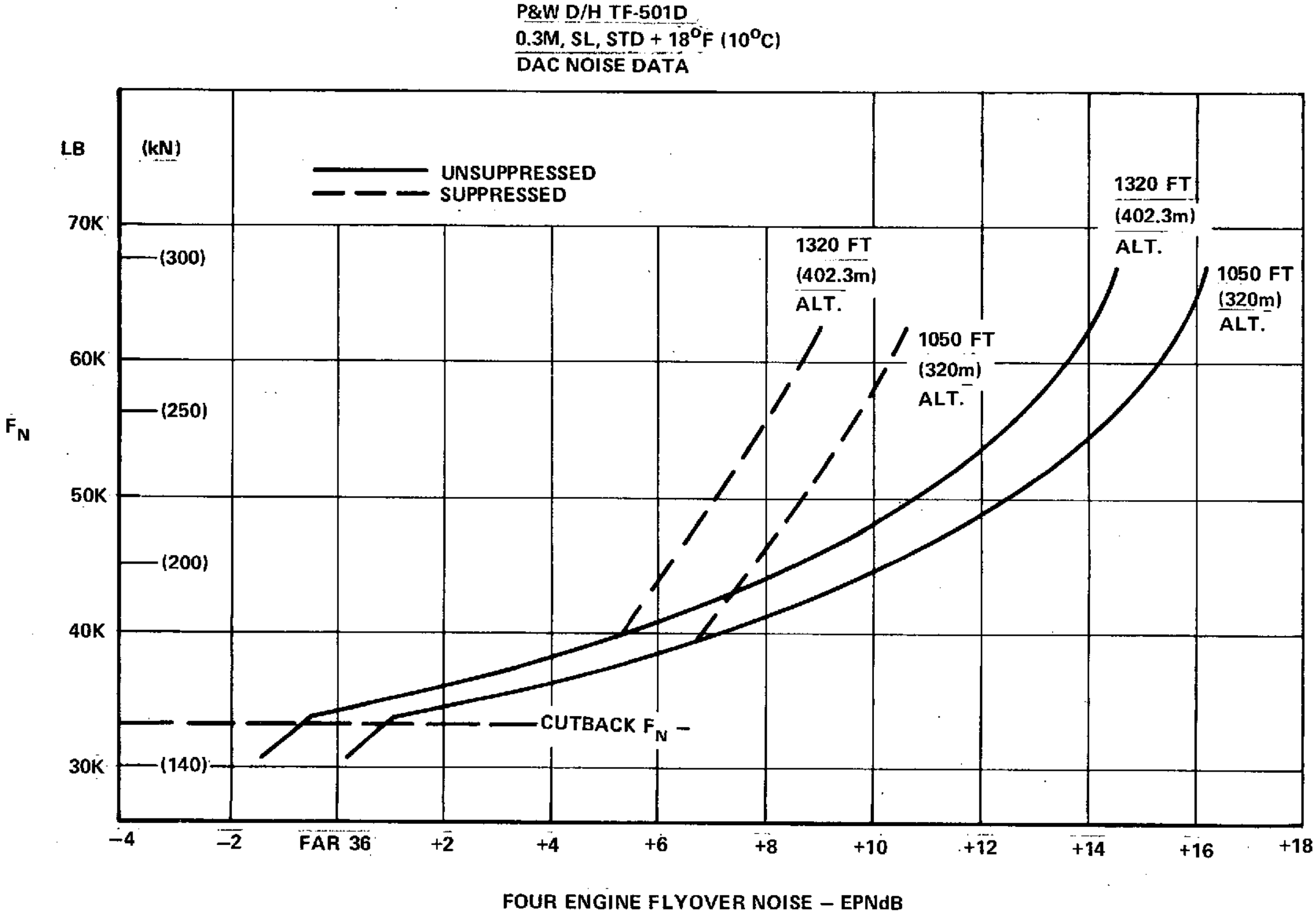


FIGURE 3-9. IN-FLIGHT NOISE CHARACTERISTICS

For this configuration, the altitude at 3.5 n.mi. (6.5 km) is 1268 ft. (386 m) (see Figure 3-24). This results in a noise value of 107.6 EPNdB, and importantly, the suppressor is stowed at and beyond this point. The 875 lb/sec (397 kg/sec) engine is the minimum size engine to meet FAR Part 36 sideline and takeoff noise requirements.

Engine Definition

The engine is a P&WA twin spool duct heating turbofan, designated the 501D, which is designed for Mach 2.2 supersonic cruise operation and incorporates 1980 technology (Figure 3-5). It incorporates a DAC designed 2.2M inlet which is sized for an inlet corrected airflow of 875 lb/sec (397 kg/sec) at takeoff rating for sea level static, Std. + 18°F (10°C) day. The design cycle characteristics and ratings are shown in Table 3-1.

The nozzle for this engine is a variable area type (variable throat and exit areas) containing an integral thrust reverser, ejector and jet noise suppressor on the fan stream. Both the primary and fan duct throat areas are variable. In an actual design a fixed primary nozzle is probably desired for design simplicity. It is assumed by P&WA that the engine cycle could be tailored to produce equivalent performance with a fixed primary exhaust control nozzle. Initial layout and sizing studies utilized the P&WA recommended chute suppressor for the -501D engine. Design layouts revealed that this type suppressor would not allow canting of the exhaust due to the length of the straight translating section. Further information from P&WA revealed that the chute suppressor had a temperature limit of 1200°F (922°K) for the size shown. In order to raise the limit to 1500°F (1089°K), the diameter over the exhaust system would have to increase by four inches (10.2 cm). Therefore, an evaluation was made of alternate suppressor schemes, provided by P&WA per DAC request. On the basis

TABLE 3-1

DUCT HEATING TURBOFAN ENGINE CHARACTERISTICS SUMMARY
875 LB/SEC (397 kg/SEC) RATED AIRFLOW

DESIGN CYCLE CHARACTERISTICS

BYPASS RATIO	2.1
FAN PRESSURE RATIO	3.3
CYCLE PRESSURE RATIO	15.0
COMBUSTOR EXIT TEMP	2700°F (1756°K) [T.O., STD + 18°F (10°C)] 2600°F (1700°K) [SUPERSONIC CLIMB STD + 18°F (10°C)]

TAKEOFF RATINGS [STD DAY + 18°F (10°C)]

MAX THRUST (SLS) – LB (kN)	70,000 (311.37)
MAX THRUST (SL, 0.3 M, UNINSTALLED) – LB (kN)	65,150 (289.8)
THRUST AT 1500°F (1089°K) EGT (SL, 0.3 M, UNINSTALLED) – LB (kN)	54,750 (243.54)

WEIGHT

ENGINE – LB (kg)	9,020 (4091.5)
ENGINE/NOZZLE/REVERSER/SUPPRESSOR – LB (kg)	12,220 (5543.0)

DIMENSIONS

ENGINE INLET GAS FLOW PATH DIAMETER – IN. (m)	74.5 (1.892)
HUB-TO-TIP RATIO (AT PLANE OF ATTACH FLANGE)	0.315
ENGINE MAX DIAMETER – IN. (m)	86.8 (2.205)
LENGTH – INLET FLANGE TO EXHAUST PLANE – IN. (m)	251.0 (6.375)

SCALING FACTORS

WEIGHT $$\frac{\text{WT}}{\text{WT BASE}} = \left(\frac{\text{WAT2}}{875}\right)^{1.085}$$

DIAMETER $$\frac{\text{D}}{\text{D BASE}} = \left(\frac{\text{WAT2}}{875}\right)^{0.5}$$

LENGTH $$\frac{\text{L}}{\text{L BASE}} = \left(\frac{\text{WAT2}}{875}\right)^{0.42}$$

COST*

WITHOUT SUPPRESSOR	\$3.01M
WITH P&WA 5.1 PNdB SUPPRESSOR	\$3.18M

SCALING FACTOR $$\frac{\text{COST}}{\text{COST BASE}} = \left(\frac{\text{WAT2}}{875}\right)^{0.53}$$

*BASED ON:
- 1973 DOLLARS
- 1980 ENGINE TECHNOLOGY
- PRICES INCLUDE ALL DEVELOPMENT COSTS PLUS FIVE-YEAR PRODUCT SUPPORT AFTER CERTIFICATION, BASED ON ONE-ENGINE MODEL
- 3000-ENGINE PRODUCTION RUN

of temperature, size, weight and installation compatibility of the various suppressors, a finger mixer system capable of 5.1 PNdB suppression at 1500°F (1089°K) with a net thrust loss of 4.6 percent has been selected as the baseline suppressor concept for this engine. The jet suppressor is designed to work on the fan stream only with the engine cycle being matched such that suppression is not required on the primary stream. The base engine including the P&WA nozzle is described in Figure 3-10. The installed engine is shown in Figure 3-11.

Engine weights, dimensions, scaling equations and cost data are presented in Table 3-1. The cost data are based on P&WA cost information provided as part of their Advanced Supersonic Propulsion System Technology Studies conducted under contract to NASA Lewis in 1973. Costs have been escalated to 1973 by DAC based on 1972 dollar values provided by the engine manufacturers' study.

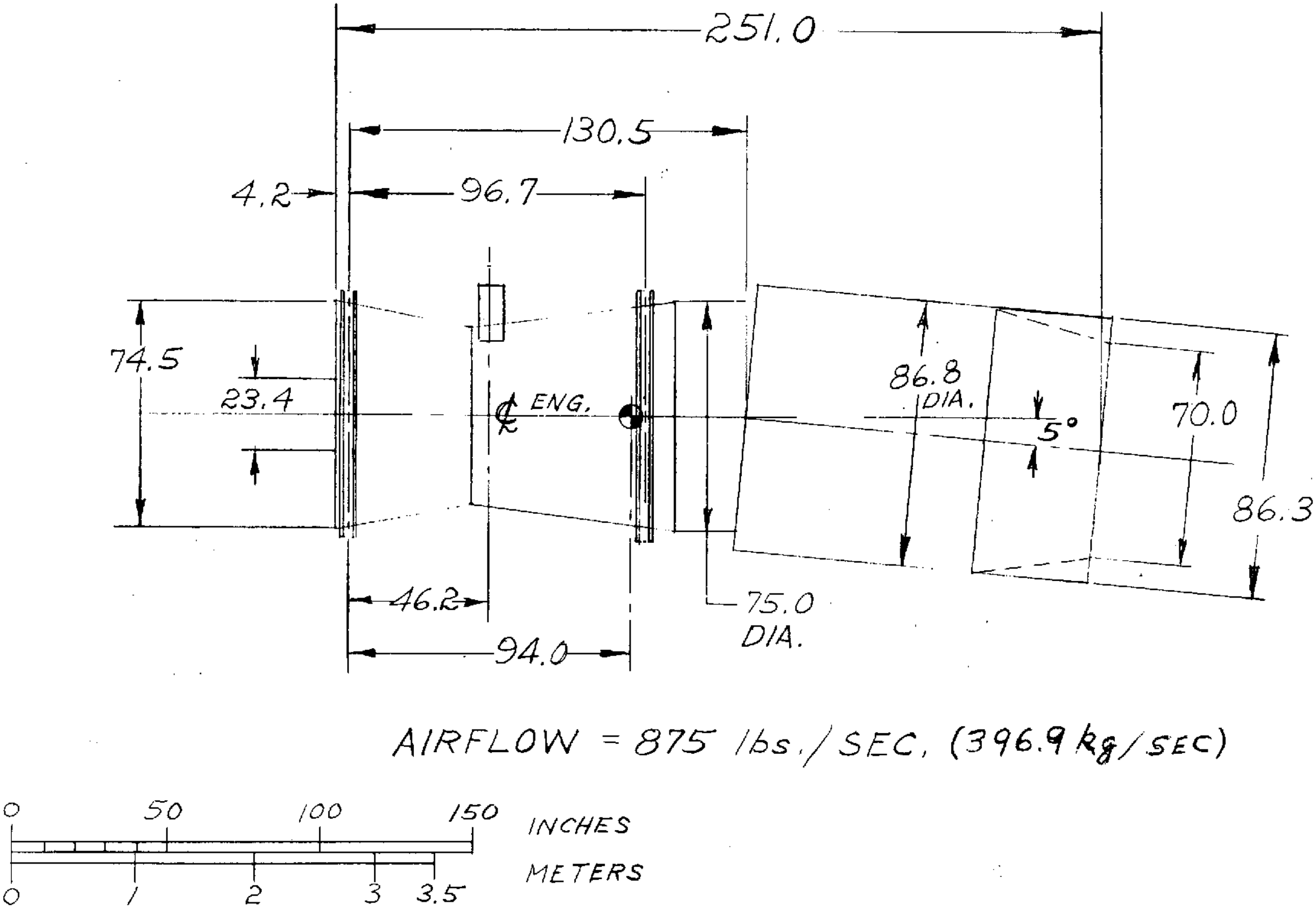
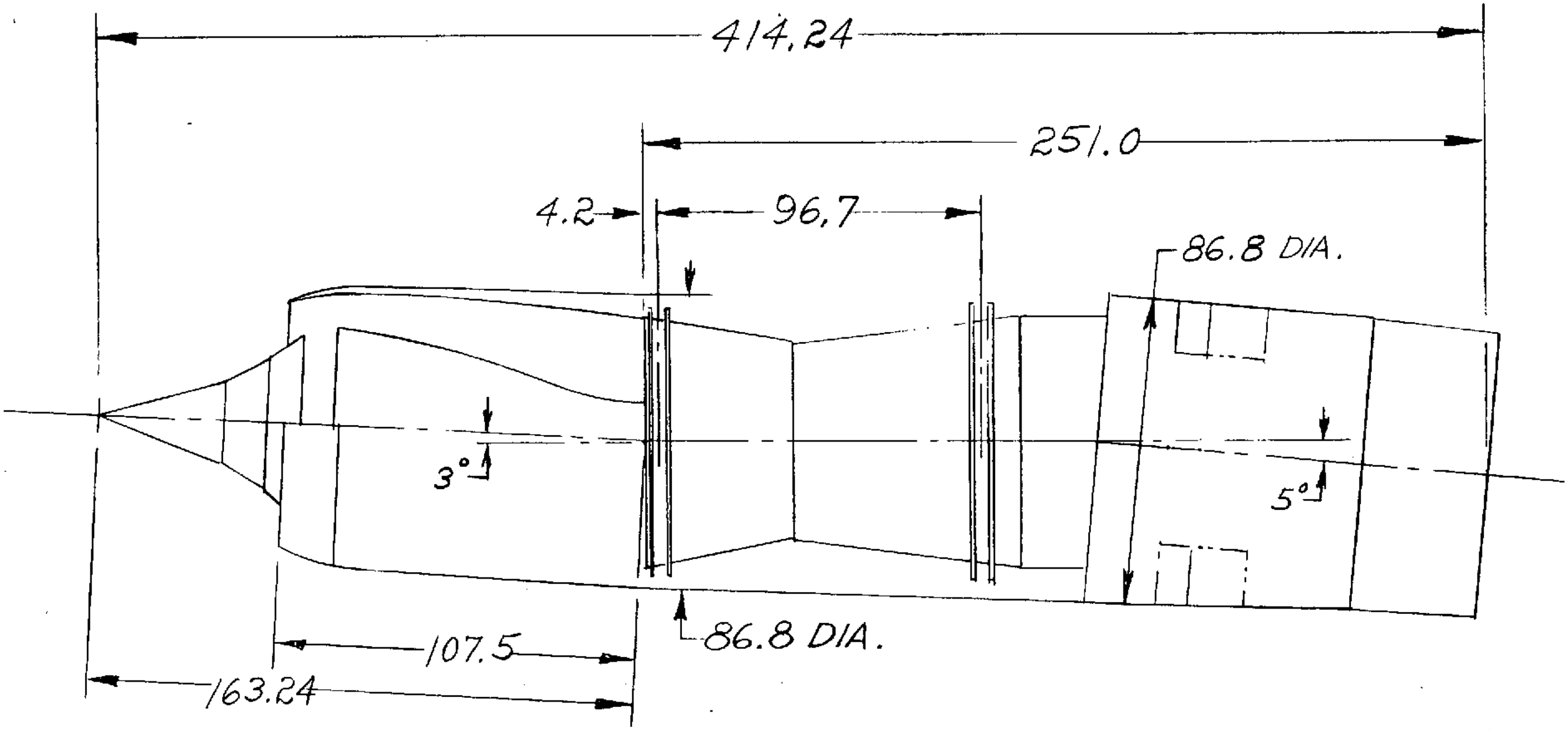


FIGURE 3-10. P&WA 501D D/H TURBOFAN ENGINE



875 LB/SEC AIRFLOW (396.9 kg/SEC)

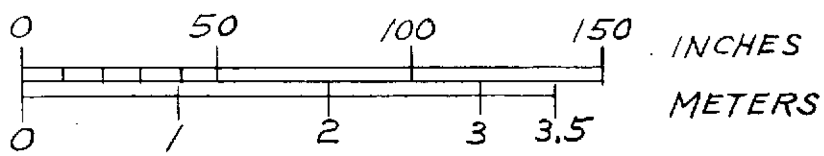


FIGURE 3-11. P&WA-501D D/H TURBOFAN ENGINE INSTALLATION

Uninstalled Performance

At DAC request, P&WA furnished uninstalled engine performance data at Mach 2.2 with DAC airflow schedule and inlet recovery. The uninstalled performance includes the effects of:

° U.S. 1962 model atmosphere	
° Inlet recovery	Figure 1-6
° P&WA supplied internal nozzle velocity coefficient	
° Customer compressor air bleed	1 lb/sec (.454 kg/sec)
° Customer power extraction	200 HP (149 kW)
° Jet A Fuel, Lower Heating Value	18,400 BTU/lb (4.34×10^7 J/kg)
° No losses for acoustical treatment	

Installed Performance Analysis

The analysis of the propulsion system performance of the duct heating engine follows the same procedures used for the baseline turbojet engine (Section 1).

The inlet performance and the nacelle analysis include an evaluation of the following items:

- Inlet spillage drag
- Inlet bypass drag
- Engine and ECS cooling airflow drag
- Nacelle skin friction drag
- Nacelle afterbody drag
- Nacelle wave drag

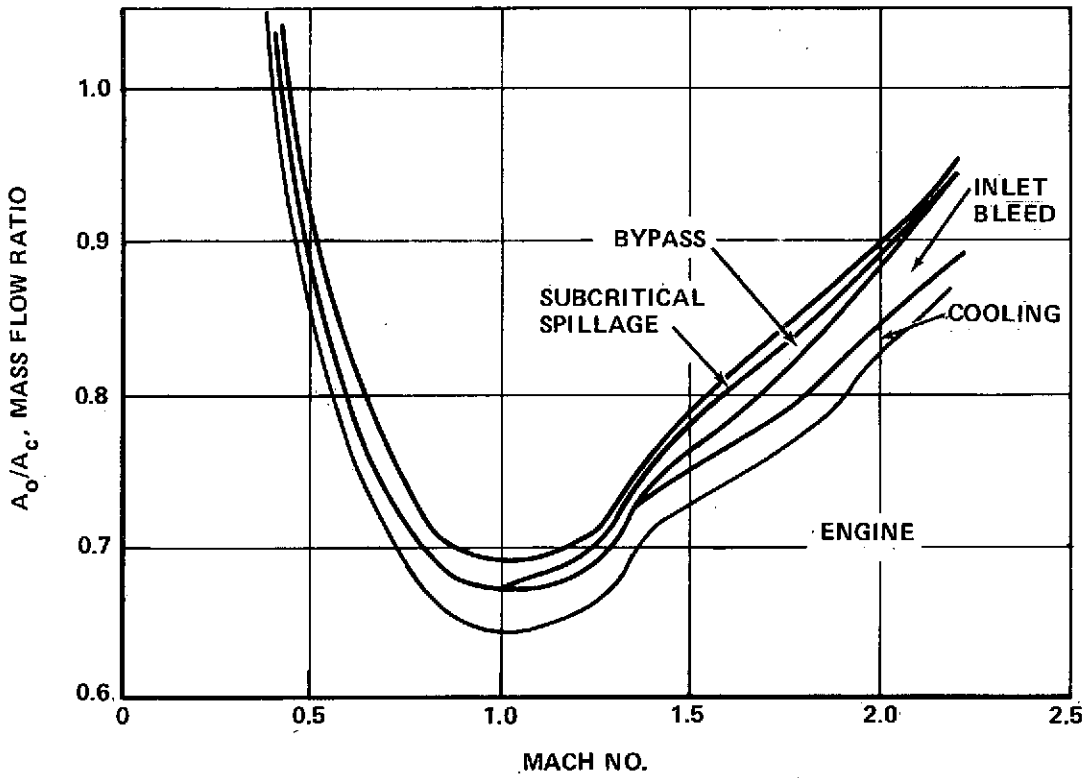
The inlet geometry and cone schedules are the same as used for the turbojet engine. The inlet total pressure recovery variation is shown in Figure 1-6. Also shown in the figure is the variation of inlet critical mass-flow ratio. Shown in Figure 1-7 is the mass-flow ratio for the inlet boundary layer bleed airflow.

The engine airflow schedule for the duct heating turbofan engine is the same as for the baseline turbojet (Figure 1-8). The installed inlet performance for the -501D engine is shown in Figure 3-12. As shown by the upper graph in the figure, the inlet airflow supply provides an adequate match with the engine airflow demand. The inlet is sized at the design point of 2.2 M. The sized capture area is 26.9 sq. ft. (2.50 sq. m.). The engine and ECS cooling airflow are based on an allowance of 2 percent of inlet capture area airflow for the environmental control system (ECS) cooling and for engine compartment ventilation and nozzle cooling.

The nacelle drag-coefficient buildup is shown in the lower graph in Figure 3-12. The inlet drag characteristics are calculated by combining the mass-flow-ratio characteristics with empirical drag coefficient correlations. For the convenience of engine sizing studies, the nacelle skin friction drag is included in the installed engine performance. The skin friction coefficients are based on fully turbulent flat plate adiabatic wall boundary layer data with transition at the leading edge. The resulting drag is shown in Figure 3-12.

The nacelle afterbody drag is dependent on the nozzle exit area and flight Mach number. The maximum nozzle area is sized at 2.2 M climb at maximum augmentation. The engine dependent boattail drag at this condition is zero. As nozzle area decreases for lower Mach numbers, and reduced power

P&WA – 501D DH/TF
A_C = 26.9 FT² (2.50 m²)



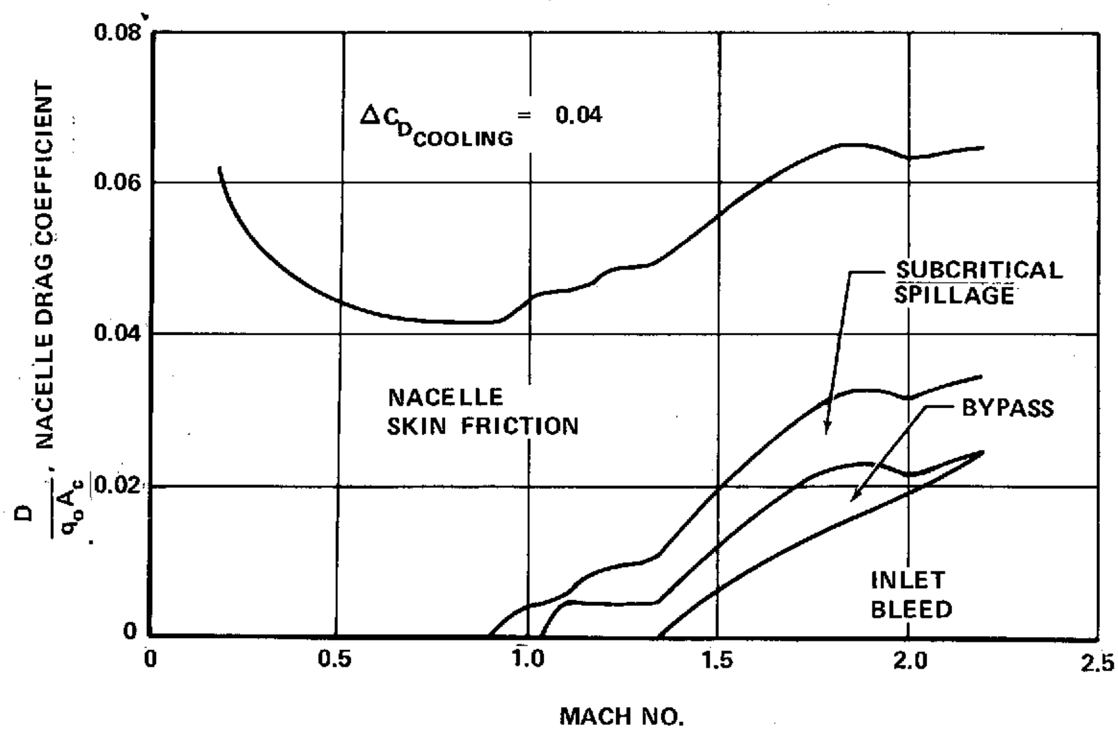


FIGURE 3-12. INSTALLED INLET PERFORMANCE

settings, the boattail drag increases. The boattail drag identified with this area change is based on drag characteristics estimated for the DAC baseline configuration. The variation in drag coefficient relative to the design cruise drag along the aircraft climb path as a function of climb thrust and for subsonic flight are shown in Figures 3-13 and 3-14.

The nacelle wave drag in the presence of the aircraft, including the supercritical spillage drag and the design afterbody drag is part of the aircraft wave drag.

Performance Results

Installed propulsion system performance is generated by correcting the uninstalled engine performance data for the installation effects described above. The climb performance characteristics are generated along the aircraft flight path shown earlier in Figure 1-12. Uninstalled and installed thrust for the takeoff power setting (EGT limited for noise) are shown in Figure 3-15. Figures 3-16 and 3-17 show the uninstalled and installed thrust and SFC, respectively, for maximum climb thrust along the climb flight path. Uninstalled and installed supersonic cruise, subsonic cruise (for alternate mission), and hold performance are shown in Figure 3-18 through 3-20. Note that the afterbody drag associated with subsonic cruise results in a significant installation penalty (see Figure 3-20). Figure 3-21 presents the installed characteristics used along the descent flight path.

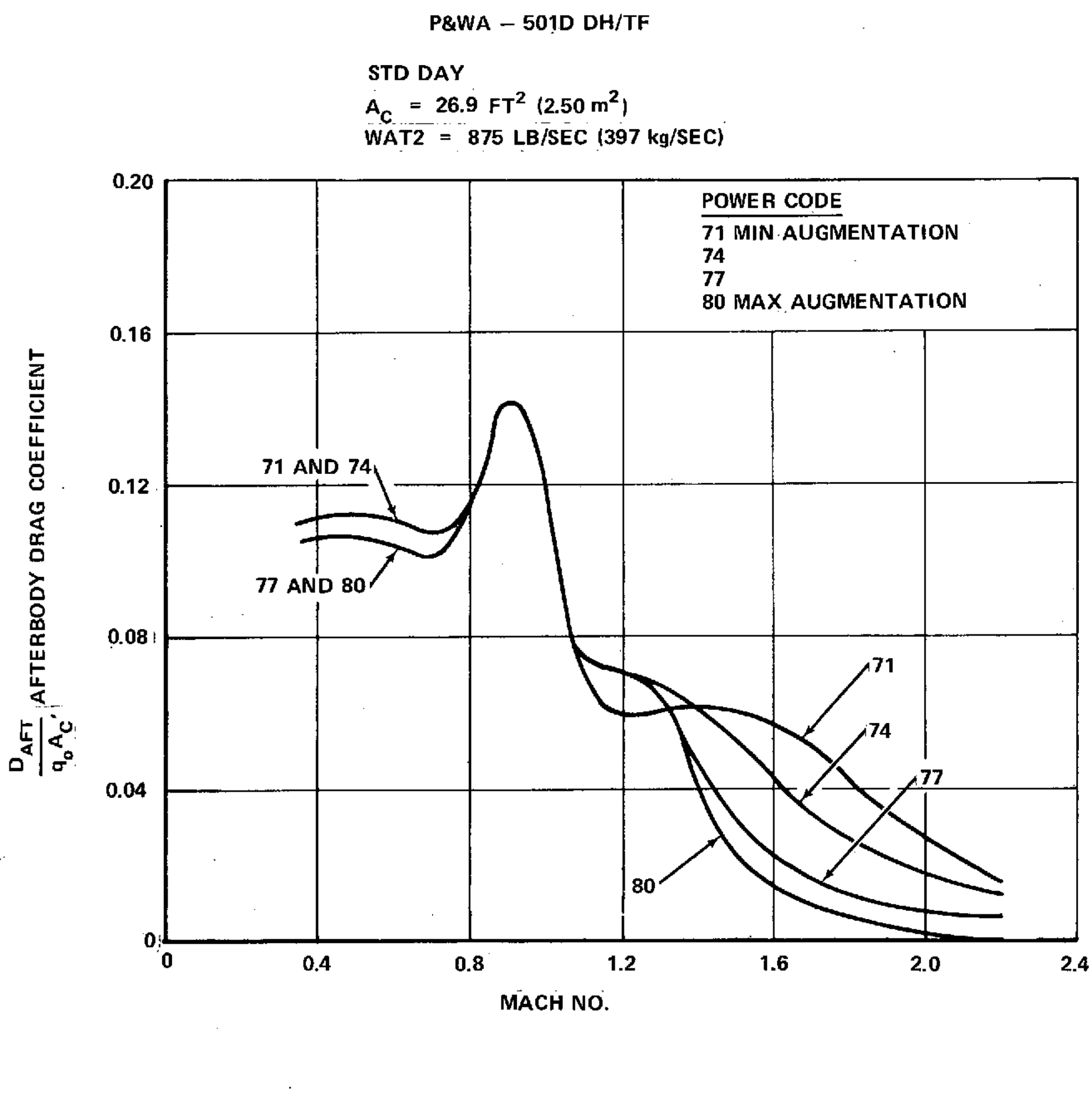


FIGURE 3-13. CLIMB AFTERBODY DRAG

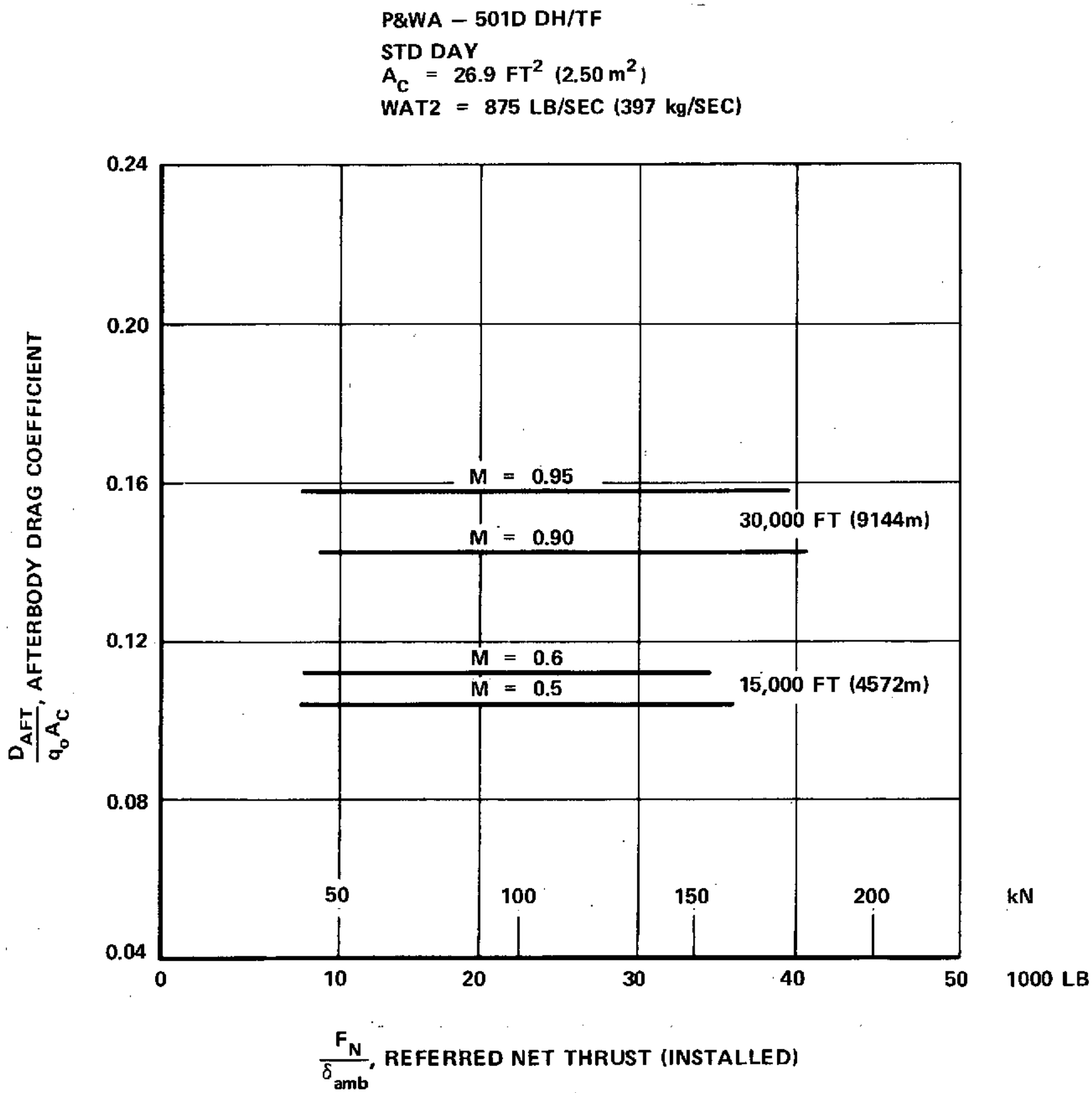


FIGURE 3-14. SUBSONIC AFTERBODY DRAG

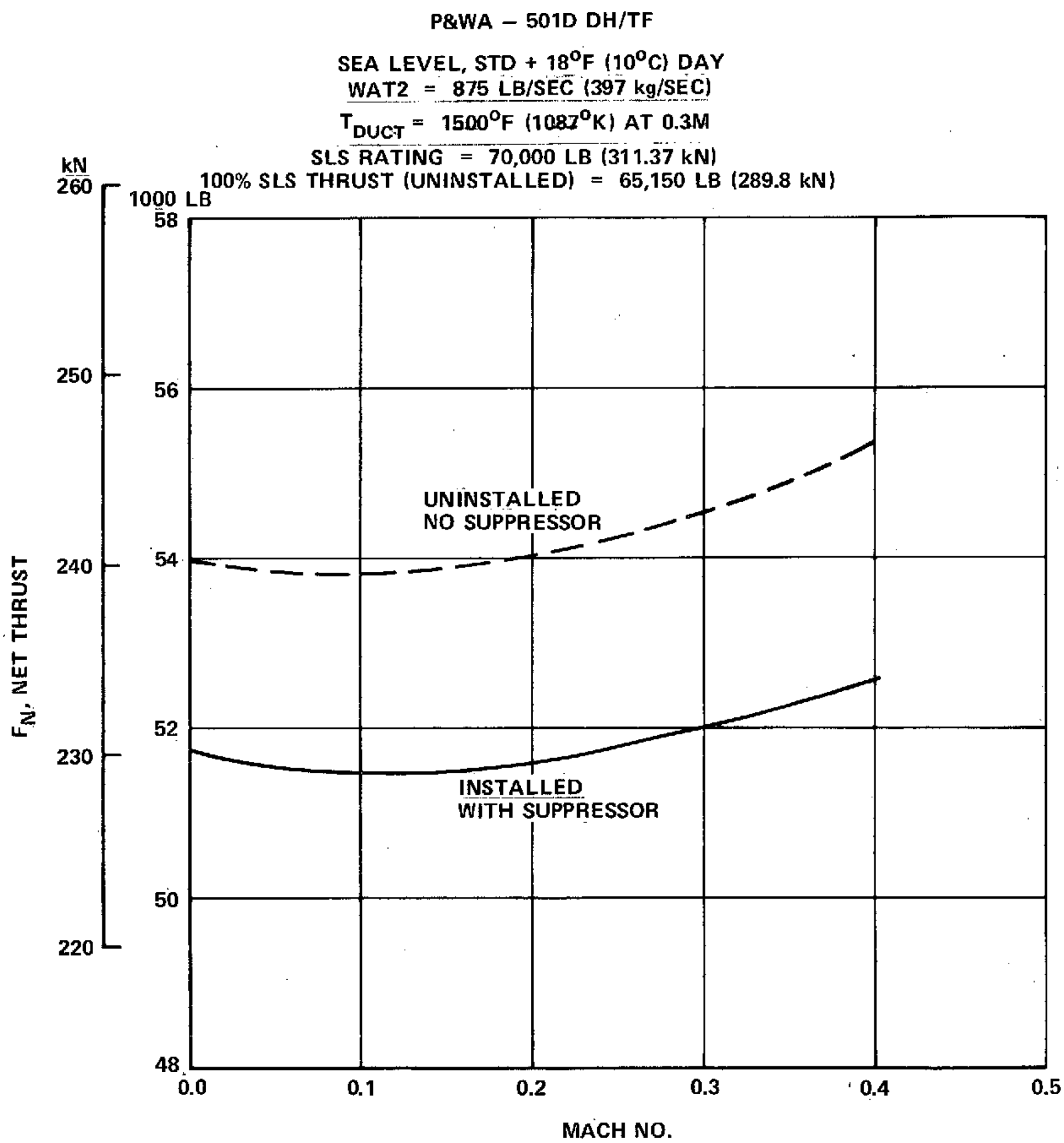


FIGURE 3-15. TAKEOFF PERFORMANCE

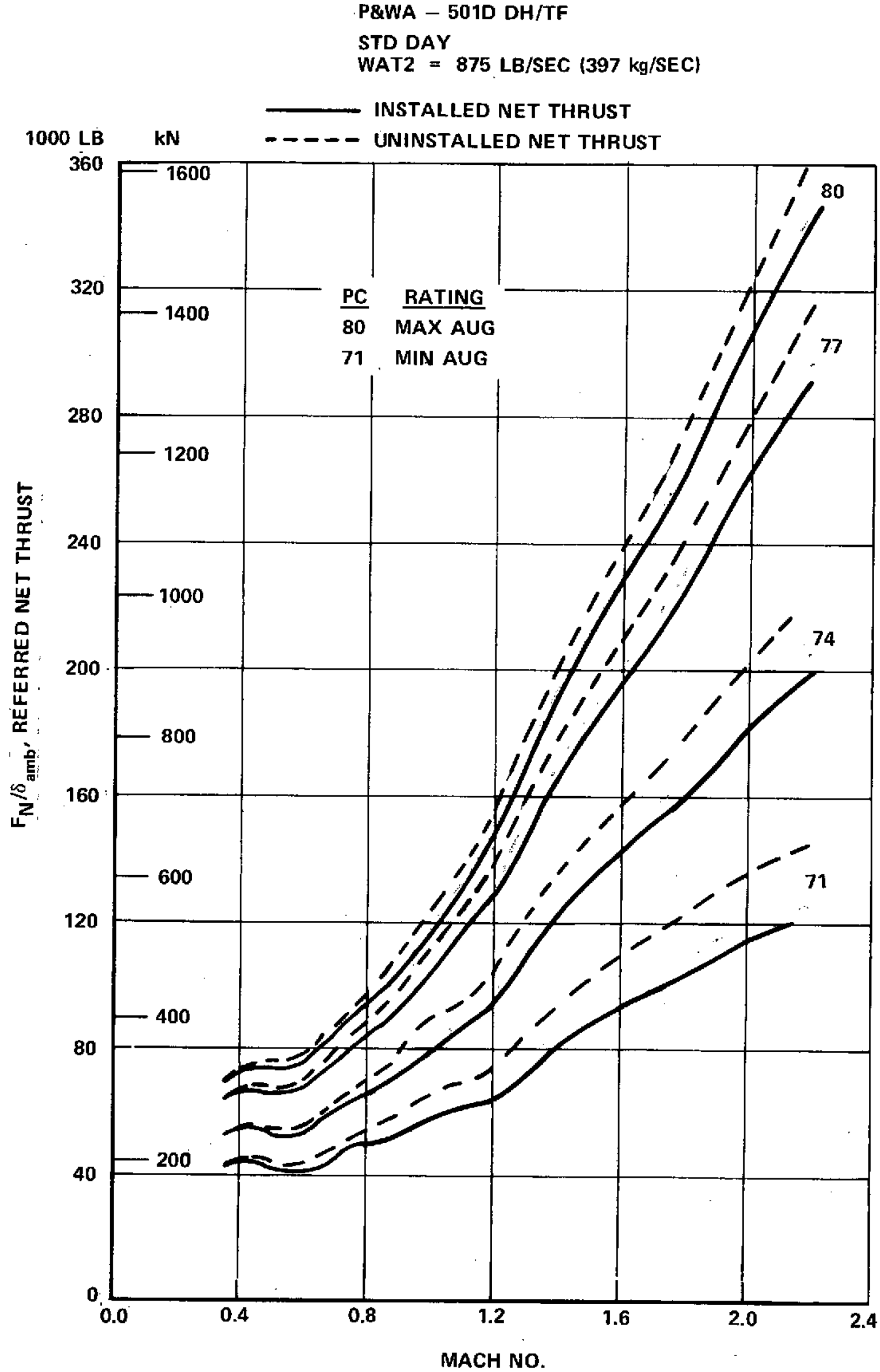


FIGURE 3-16. CLIMB THRUST

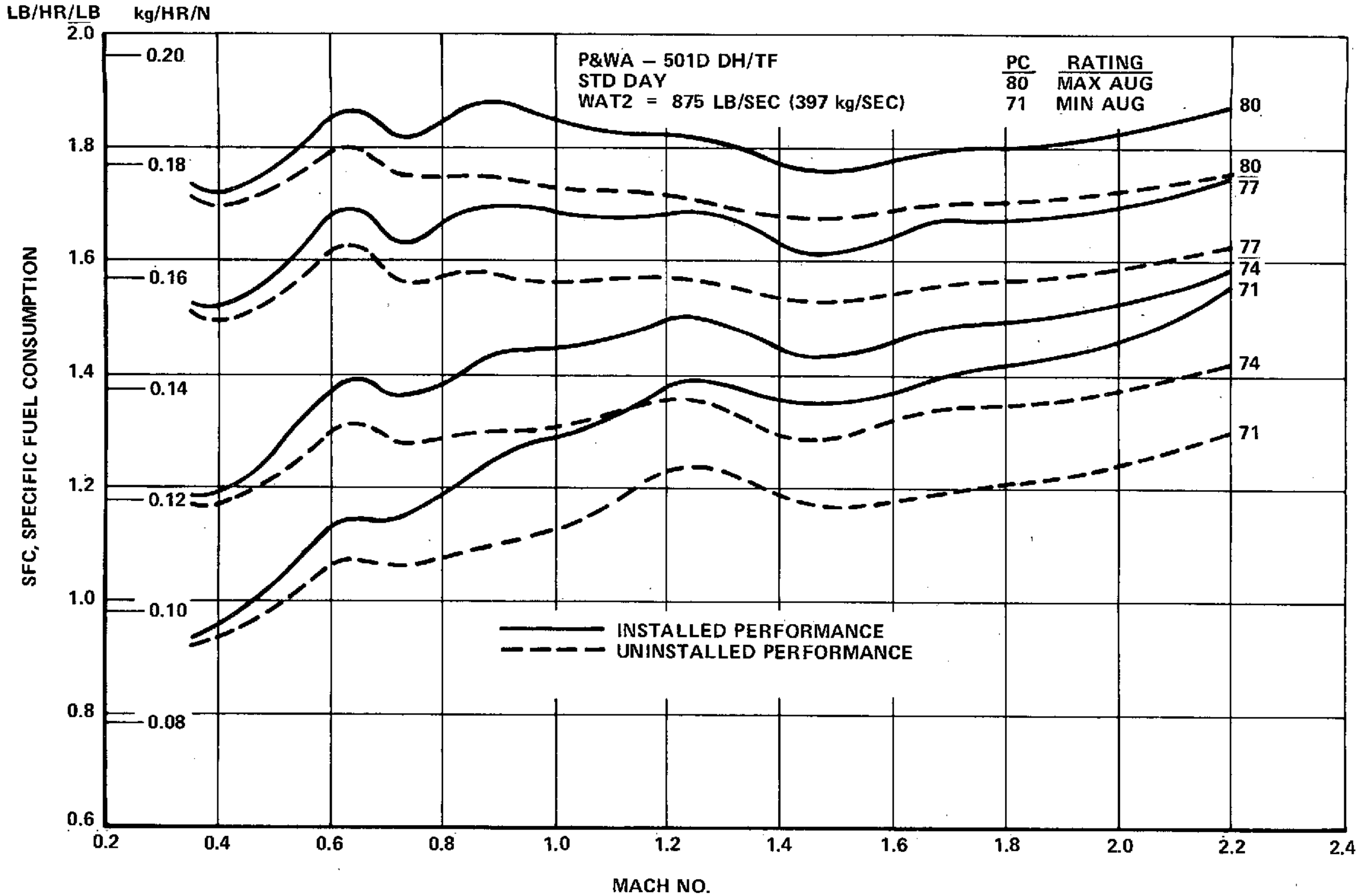


FIGURE 3-17. CLIMB SFC

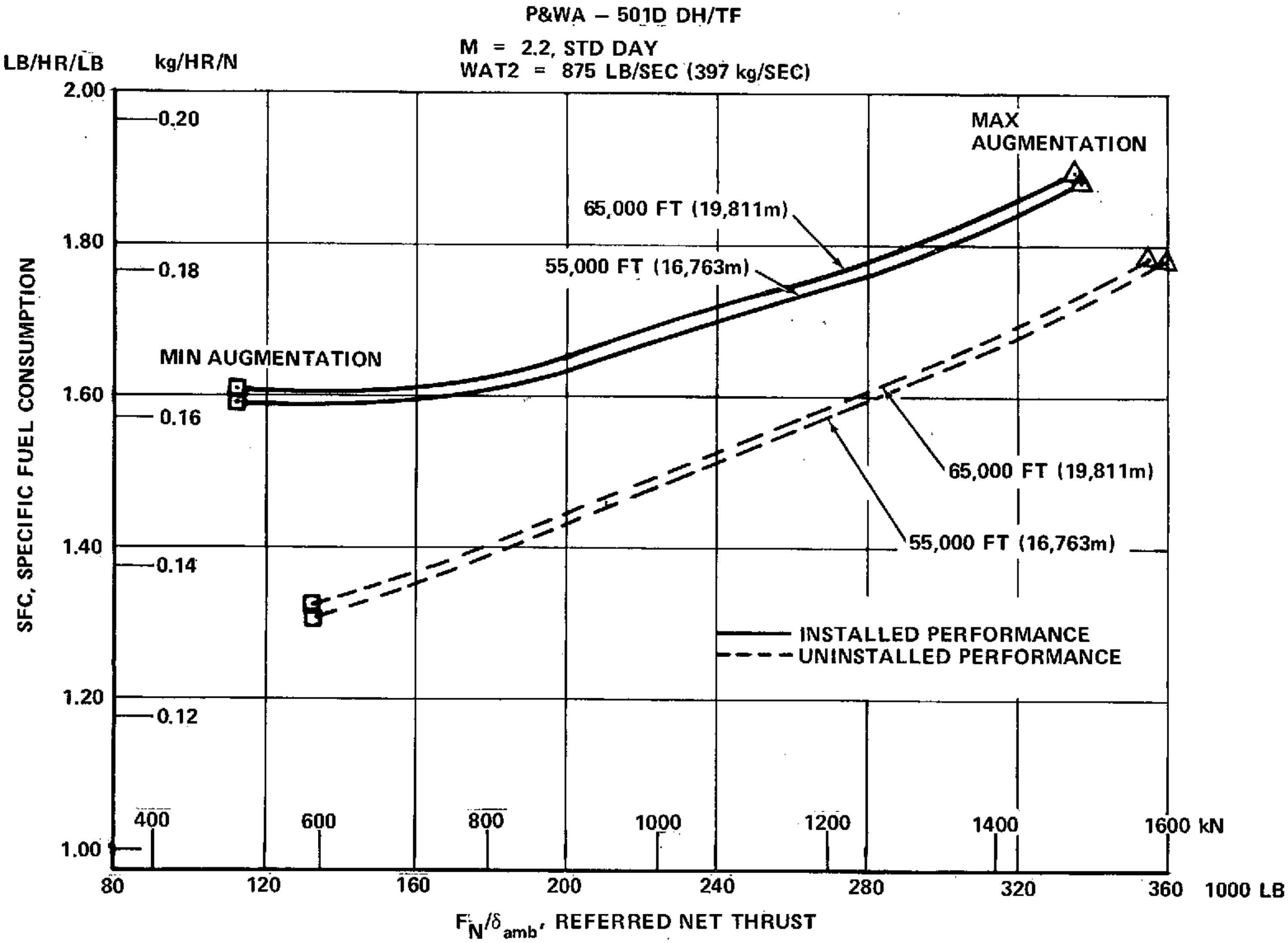


FIGURE 3-18. SUPERSONIC CRUISE PERFORMANCE

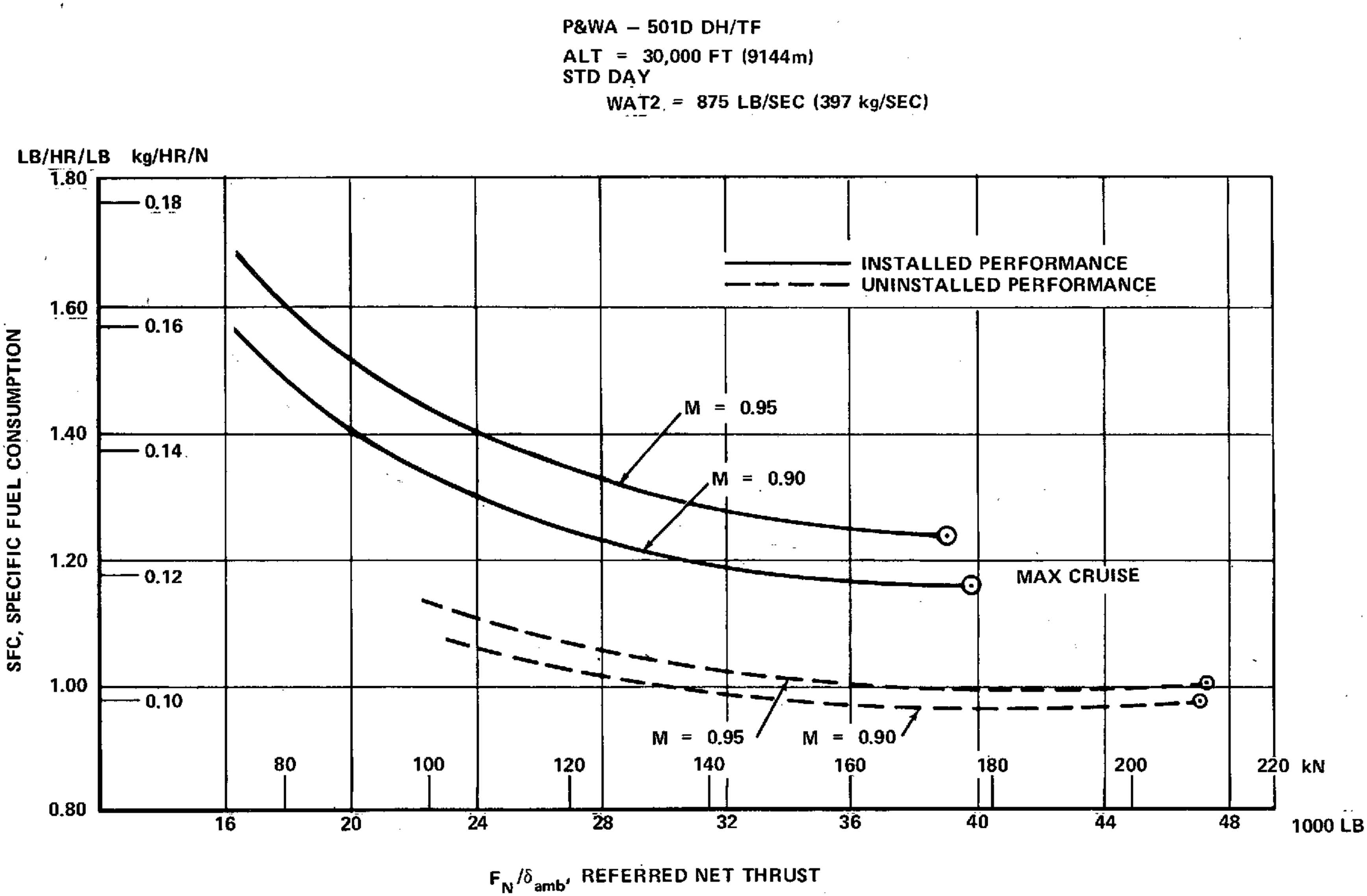


FIGURE 3-19. SUBSONIC CRUISE PERFORMANCE

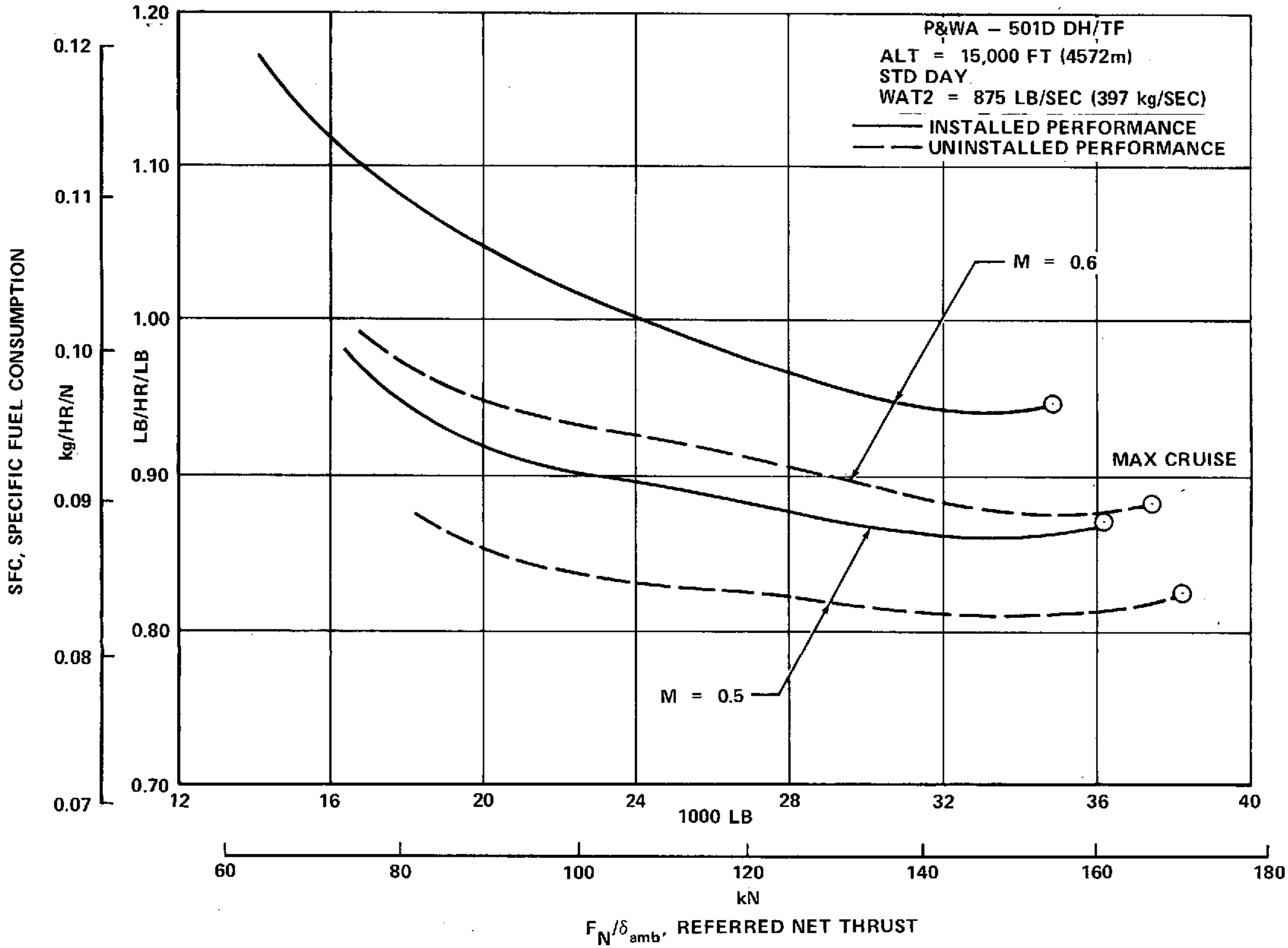


FIGURE 3-20. LOITER PERFORMANCE

P&WA – 501D DH/TF
STD DAY
WAT2 = 875 LB/SEC (397 kg/SEC)

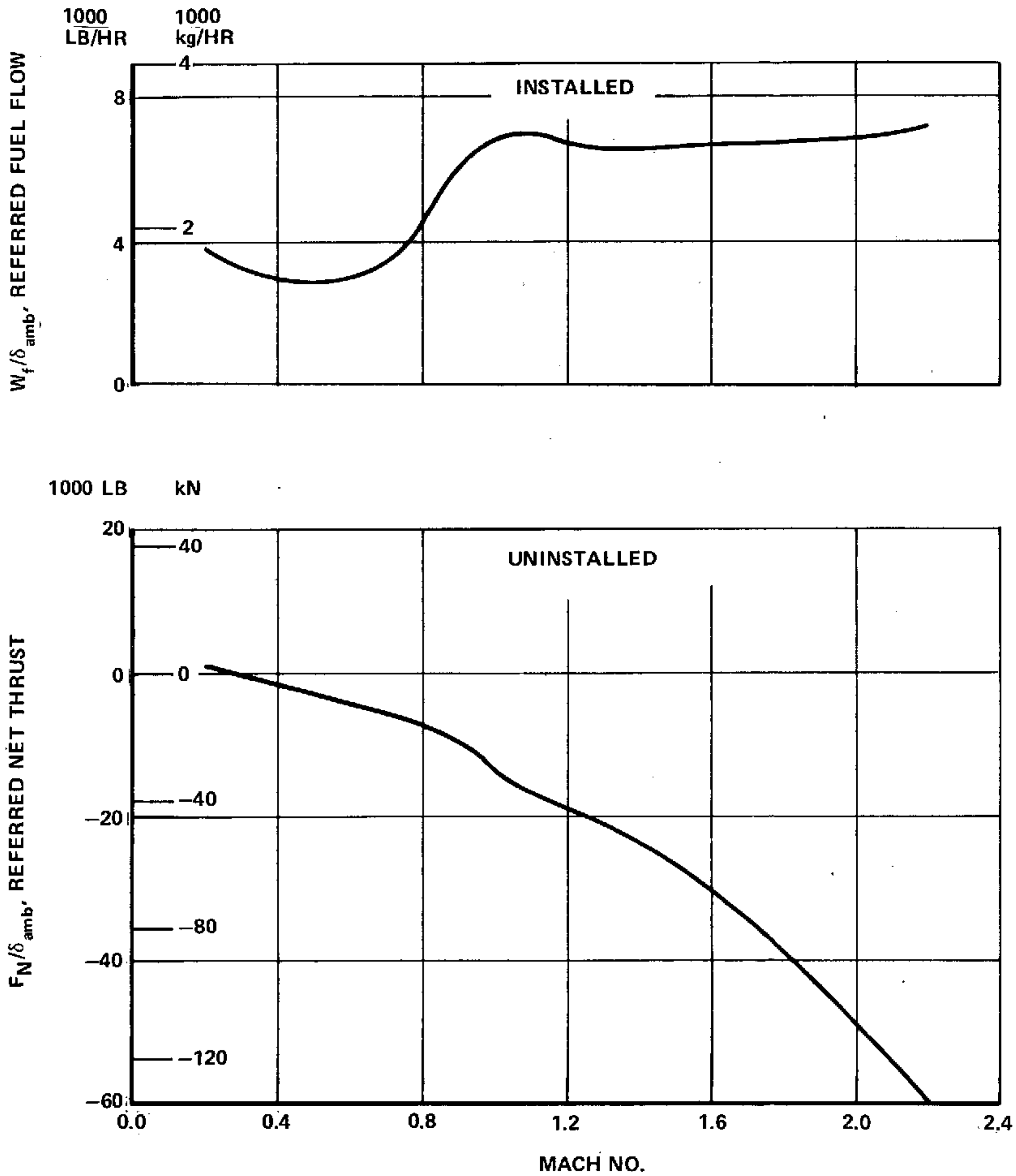


FIGURE 3-21. IDLE PERFORMANCE

Engine/Nacelle Location

Installation studies of the duct heating turbofan engines for the baseline airframe in four axisymmetric nacelles have been completed. Inboard and outboard spanwise locations remain as for the -5A and -5B configurations to maintain existing wing torque box structure, disposition of control surfaces, and overall area distribution equivalent to the baseline configuration. The choice of the forward and aft location has been determined from the inputs of the aerodynamics, structural mechanics, acoustics, and power plant groups.

With the resultant location of engine to wing, use of the 360° reverse thrust nozzles proposed by the engine manufacturer cannot be utilized. Thrust reversal is only achievable in local areas (70°) above the upper wing surface and (150°) between the deployed aircraft flap system (Figure 3-22).

The locations as shown on the three view drawing provide the best solution to the requirements of the previously established criteria (Figure 3-23).

Engine/Nacelle Attachment to Wing

The engine is mounted on the wing by a three point attachment to the wing structure. The aft mount is attached to a box beam pylon cantilevered aft of the rear spar and the two forward mounts are attached to structure provided on the wing box off the rear spar.

The forward right hand mount carries thrust loads, vertical loads and side loads. The left hand forward mount transmits forward and vertical loads only.

The rear engine mount carries vertical loads and translates for engine growth under operating temperatures (see Figure 3-22).

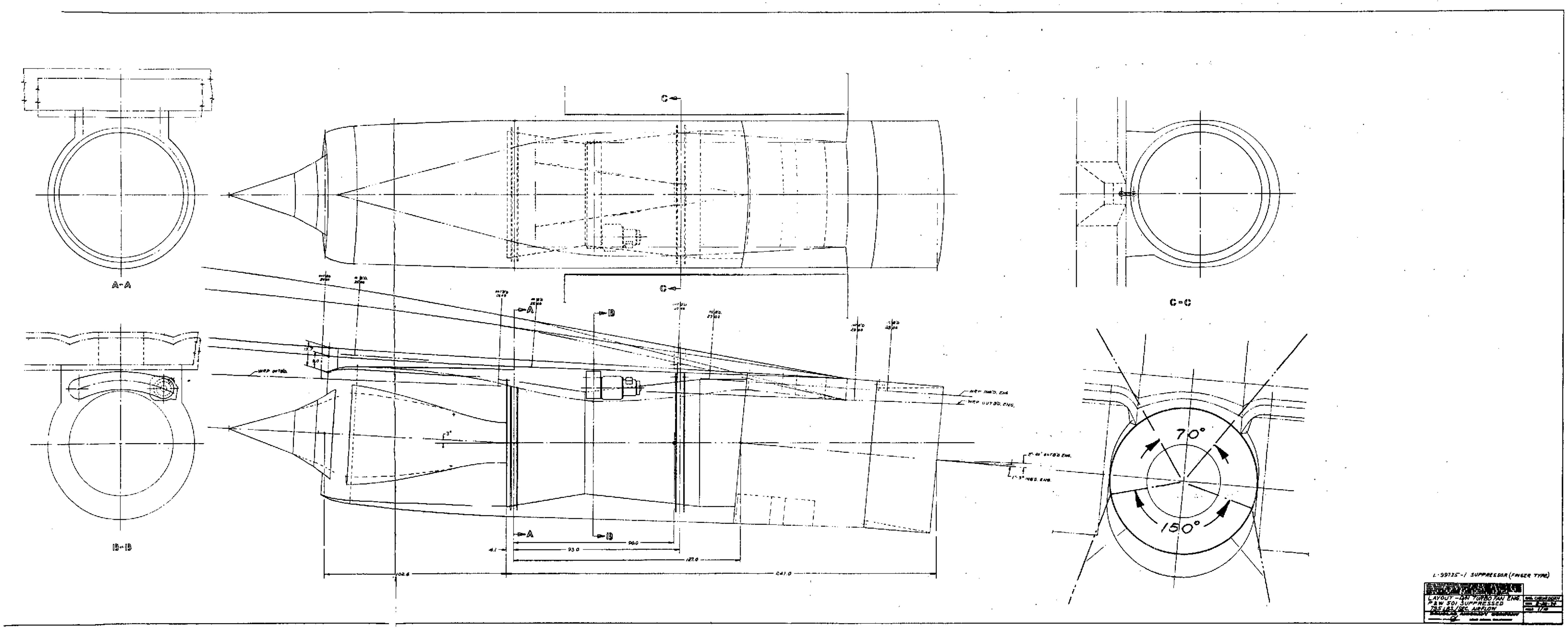


FIGURE 3-22. P&WA 501D ENGINE INSTALLATION SCHEMATIC

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FOLDOUT FRAME

FOLDOUT FRAME

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CHARACTERISTICS			
ITEM	WING	HORIZONTAL STABILIZER	VERTICAL STABILIZER
AREA SQ.FT.	10000	780	700
ASPECT RATIO	1.84	2.0	0.9
TAPER RATIO	.128 (TRAP)	0.16	0.25
SWEEP OF L.E.	71° & 57°	50°	50°
DIHEDRAL	0 AT T.E.	0	~
T/C %C.	2.25 - 3.0	3.5	3.5
TAIL VOL. COEF.	~	.118	.044

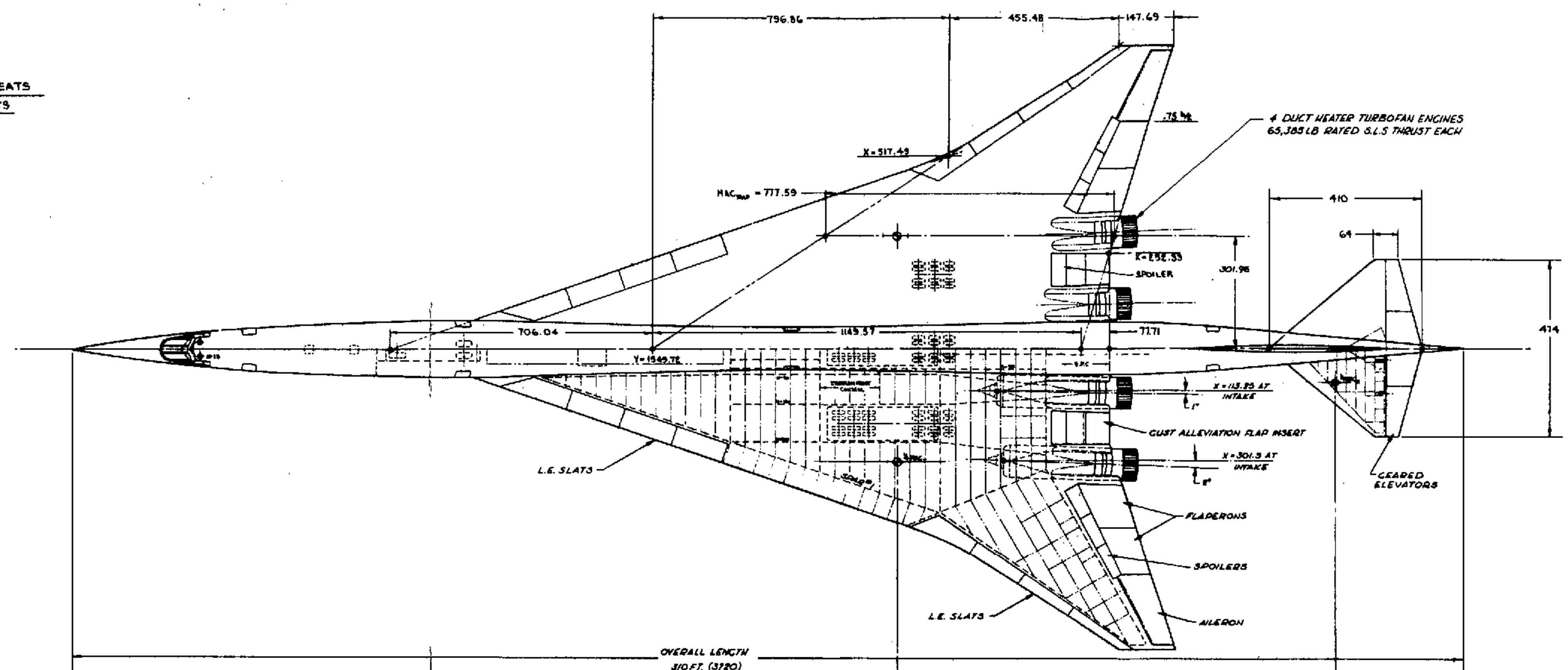
PAYLOAD CAPACITY

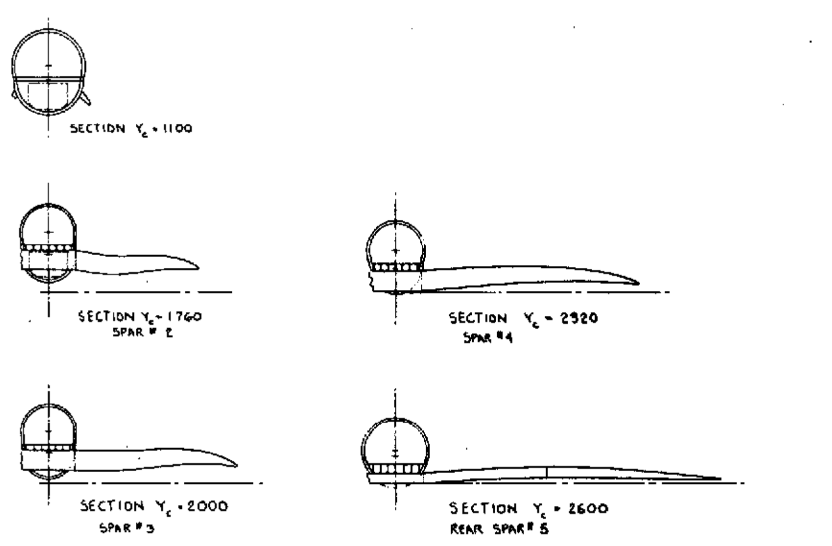
MIXED CLASS

FIRST CLASS – 3 & 4 ABREAST – 38 PITCH – 40 SEATS
TOURIST CLASS – 4 5 & 6 ABREAST – 34 PITCH – 235 SEATS
TOTAL 275 SEATS

CARGO VOLUME

FORWARD	1500 CU. FT.
AFT	750 CU. FT.
TOTAL	2250 CU FT





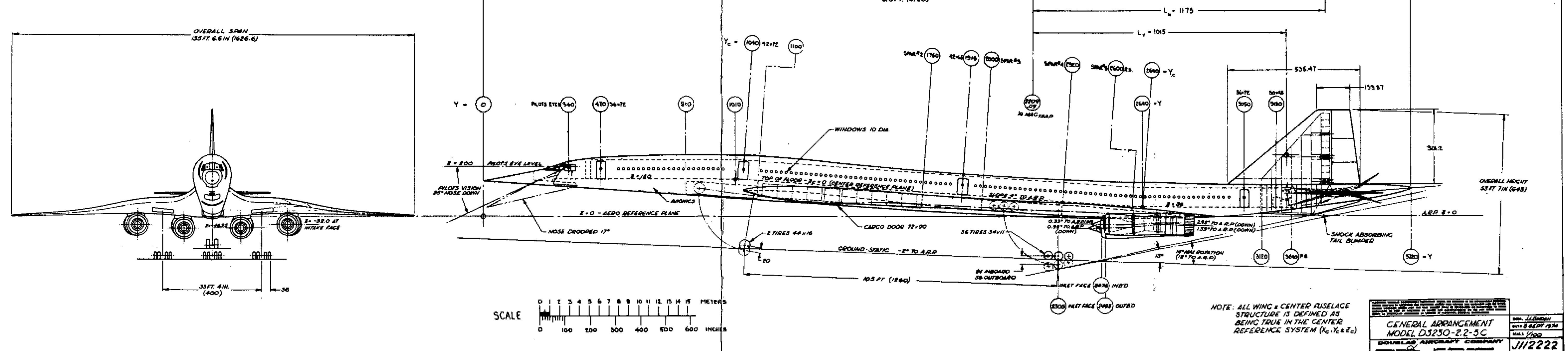


FIGURE 3-23. AST P&WA DUCT HEATING TURBOFAN ENGINE CONFIGURATION

The axisymmetric intakes are mounted to the engine casing and divorced from the wing structure. This eliminates transmission of wing deflection loads to the intake to prevent distortion of the intake geometry and loading of the engine casing.

The boundary layer diverter is integrated into the engine nacelle/wing fairing.

The installed weight of the duct heating turbofan is less than either the baseline turbojet or mini-bypass engines, which results in significant weight savings in airframe structures.

Other Configuration Changes

The reduced length of the installed engine pods for the duct heating turbofan engines enables the use of the shorter landing gear, reduced tail bumper fairing, and the ground clearances established for the -5B configuration aircraft.

ACOUSTIC ANALYSIS

The acoustic analysis conducted for the duct heating turbofan powered aircraft configuration consists of the calculation of estimated jet noise in support of engine sizing studies. Engine data have been employed to estimate the jet noise at aircraft Mach numbers and altitudes representative of the FAR Part 36 takeoff and sideline measuring conditions. After the engine size has been determined, the flight path for the AST duct heating turbofan powered aircraft configuration is calculated and engine cycle data at the above two conditions defined. The standard climb profile featured a thrust cutback over the takeoff measuring station.

The engine size has been selected at an airflow rate of 875 lbs/sec (397 kg/sec) in combination with a jet noise suppression in the fan duct designed to provide a nominal 5.1 PNdB sideline suppression. Descriptions of the engine sizing results and of the selected jet noise suppressor are given in the Engine Sizing section.

The unsuppressed jet noise levels for the -501D engine in the baseline airplane based on specific engine conditions on the calculated takeoff trajectory are as follows:

FAR PART 36 MEASURING STATION	DISTANCE, FT.(m)	UNSUPPRESSED TOTAL NOISE, EPNL, EPNdB
Sideline	2270 (747)	116.1*
Takeoff / cutback	1268 (386)	108.1

*Includes no allowance for extra ground attenuation

The suppressed noise levels for this configuration are estimated as described in the Engine Sizing section.

Strength Analysis

The -501D engine propulsion system plus nacelle weight from Table 3-2 is 66,709 lbs. (30,259 kg). This compares to 84,920 lbs. (38,519 kg) for the baseline -5A. Therefore, the weight reduction is 9106 lbs. (4130 kg) per side. The saving in structural weight for this decrease in propulsion weight is 670 lbs. (304 kg) per side (point 7 minus point 6 in Figure 4-27). The 23.5 inch (60 cm) more forward location for the duct heating turbofan engine c.g. results in an additional 400 lbs. (181 kg) per side structural weight saving (point 6 in Figure 4-27). This totals 1070 lbs. (485 kg) per side or 2140 lbs. (971 kg) per airplane structural weight saving.

Flutter Analysis

A flutter analysis of the -501D engine configuration (-5C), revealed that this light weight engine was the most critical case as far as meeting the flutter requirements (see further discussion in Section 4). The 860 lb. (390 kg) allocation for detailed flutter optimization is anticipated to satisfy flutter requirements for the -501D engine (see Table 3-2).

Table 3-2 compares the weight of the AST with -501D engines (-5C) to the turbojet baseline (-5A). Bare engine weight of the P&WA model D/H TF-501D is 9,020 lbs. (4091 kg) each. The nozzle, reverser and suppressor are an additional 3,200 lbs. (1451 kg). This compares to 12,942 lbs. (5870 kg) and 4,040 lbs. (1833 kg), respectively, for the baseline -5A. Total propulsion system weight is 51,057 lbs. (23,159 kg), 19,133 lbs. (8679 kg) less than the turbojet baseline, 8,874 lbs. (4025 kg) less than the mini-bypass configuration -5B.

In contrast, the weight of the nacelle/inlet is 922 lbs. (418 kg) heavier than the baseline -5A. This is a net weight change, comprised of a 751 lb. (341 kg) reduction in the weight of engine cowling and a 1,673 lb. (759 kg) increase in the weight of the engine inlet. The decrease in cowling weight reflects an engine envelope approximately 32 inches (81 cm) shorter and 10 inches (25 cm) less in diameter. The heavier inlet reflects increases in both capture area and duct length.

The Structural Weight Increment includes differences in pylon and engine support weight, along with differences in wing and fuselage weight, due to changes in load. The 2,140 lbs. (971 kg) reduction (see Structural Analysis paragraph), estimated for this increment, is based on results from structural optimization studies of both the baseline and mini-bypass configurations. Figure 4-27 presents the results from an engine location study of the baseline configuration (-5A). These data, combined with the results from the structural optimization of model -5B, provide a variation of structural weight with engine location and installed weight. This approach, details of which are presented in Section 4, is used to estimate the structural weight increment for both the duct heating turbofan and variable cycle engine concepts.

TABLE 3-2.

WEIGHT COMPARISON – CONFIGURATION 5C (DUCT HEATER TURBOFAN) WITH 5A BASELINE (TURBOJET) ENGLISH UNITS

CONFIGURATION	WEIGHT – POUNDS		
	5A TURBOJET	5C DUCT HEATER	DIFF.
ITEM			
WING	75,347	75,245*	–102
H-TAIL	3,960	3,960*	
V-TAIL	3,807	3,807*	
FUSELAGE	47,713	47,689*	–24
LANDING GEAR	36,792	36,135	–657
FLIGHT CONTROLS	9,115	9,115	
NACELLE/INLET	14,730	15,652	+922
PROPULSION (LESS FUEL SYSTEM)	70,190	51,057	–19,133
FUEL SYSTEM	3,820	3,820	0
EMERGENCY POWER UNIT	950	950	0
INSTRUMENTS	1,227	1,227	0
HYDRAULICS	5,684	5,684	0
PNEUMATICS	1,332	1,332	0
ELECTRICAL	4,850	4,850	0
NAVIGATION AND COMMUNICATIONS SYSTEM	2,756	2,756	0
FURNISHINGS	24,478	24,478	0
AIR CONDITIONING	4,854	4,854	0
ICE PROTECTION	489	489	0
HANDLING PROVISIONS	90	90	0
PENALTY – FLUTTER AND AEROELASTICITY	2,860**	2,860**	0
STRUCTURAL WEIGHT INCREMENT	––	–2,140*	–2,140
MANUFACTURER'S EMPTY WEIGHT (MEW)	315,044	293,910	–21,134
OPERATIONAL ITEMS	8,096	8,096	0
OPERATIONAL EMPTY WEIGHT (OEW)	323,140	302,006	–21,134

*THE WEIGHT INCREMENT FOR STRENGTH, ETC., FOR THESE ITEMS IS INCLUDED UNDER THE ITEM STRUCTURAL WEIGHT INCREMENT AND LISTED SEPARATELY.

**2000 LB FOR ROLL AND CONTROL EFFECTIVENESS
860 LB FOR FLUTTER OPTIMIZATION

TABLE 3-2.

WEIGHT COMPARISON – CONFIGURATION 5C (DUCT HEATER TURBOFAN) WITH 5A BASELINE (TURBOJET) METRIC UNITS

CONFIGURATION	WEIGHT – KILOGRAMS 5A TURBOJET	5C DUCT HEATER	DIFF.
ITEM			
WING	34,177	34,131*	–46
H-TAIL	1,796	1,796*	
V-TAIL	1,727	1,727*	
FUSELAGE	21,642	21,631*	–11
LANDING GEAR	16,689	16,391	–298
FLIGHT CONTROLS	4,134	4,134	
NACELLE/INLET	6,681	7,099	+418
PROPULSION (LESS FUEL SYSTEM)	31,838	23,160	–8678
FUEL SYSTEM	1,733	1,733	0
EMERGENCY POWER UNIT	431	431	0
INSTRUMENTS	557	557	0
HYDRAULICS	2,578	2,578	0
PNEUMATICS	604	604	0
ELECTRICAL	2,200	2,200	0
NAVIGATION AND COMMUNICATIONS SYSTEM	1,250	1,250	0
FURNISHINGS	11,103	11,103	0
AIR CONDITIONING	2,202	2,202	0
ICE PROTECTION	222	222	0
HANDLING PROVISIONS	41	41	0
PENALTY – FLUTTER AND AEROELASTICITY	1,297	1,297	0
STRUCTURAL WEIGHT INCREMENT	––	–971*	–971
MANUFACTURER'S EMPTY WEIGHT (MEW)	142,902	133,316	–9586
OPERATIONAL ITEMS	3,672	3,672	0
OPERATIONAL EMPTY WEIGHT (OEW)	146,574	136,988	–9586

*THE WEIGHT INCREMENT FOR STRENGTH, ETC., FOR THESE ITEMS IS INCLUDED UNDER THE ITEM STRUCTURAL WEIGHT INCREMENT AND LISTED SEPARATELY.

The decreased gear length due to the smaller engine is the same as for the mini-bypass configuration. Corresponding savings in gear and gear door weight are identical; 657 lbs. (298 kg) and 126 lbs. (57 kg), respectively. The weight penalty for flutter and aeroelasticity is 2,860 lbs. (1297 kg). This penalty is derived as a part of the structural/weight optimization analysis. Details of this analysis are presented in Section 4.

The mean inlet location of the -501D engines is 15.5 inches (40 cm) forward of the turbojet baseline (Sta. 2484.5 versus Sta. 2500). This, combined with the shorter engine package, moves the c.g. of the engine installation 23.5 inches (60 cm) forward. Combining the forward c.g. shift of the engine installation with the moment change due to the lighter weight [66,709 lbs. (30,259 kg) versus 84,920 lbs. (38,519 kg)] moves the OEW c.g. of the airplane 30.6 inches (78 cm) forward (the engines are located aft of the aircraft c.g.). Adding the effect of wing, gear and fuselage weight saving, moves the OEW c.g. another 2.5 inches (6.4 cm) in the same direction, for a total forward shift of 33.1 inches (84.4 cm). Total OEW saving for the duct burning turbofan engine configuration is 21,134 lbs. (9586 kg) relative to the -5A.

Aerodynamics Analysis

The trimmed lift and drag characteristics for the duct heating turbofan powered aircraft are obtained by adjusting the wave drag of the baseline turbojet powered aircraft for the difference due to the duct heating turbofan nacelles. The difference in nacelle skin friction drag is accounted for in the installed propulsion system performance. The wave drag program predicts a reduction in supersonic wave drag of 3.61 counts (ΔC_D =.000361) due to the differences in nacelle shape and location. The characteristics used to determine the mission performance for the duct heating turbofan powered aircraft are obtained by subtracting this increment from the wave drag of the baseline turbojet powered aircraft.

Performance Results

Estimated performance characteristics for the duct heating turbofan powered aircraft are presented in Figures 3-24 through 3-26 as a function of engine size. The mission profile and reserve ground rules are the same as used for the baseline turbojet aircraft (Figure 1-20). The takeoff gross weight is held constant at 750,000 lbs. (340,194 kg) and the payload is fixed at 55,965 lb. (25,385 kg).

Figure 3-24 presents the takeoff characteristics and the height above the runway at 3.5 n.mi. (6.5 km) from the start of takeoff, with the throttle cut back to meet the 4 percent all-engine climb gradient requirement of FAR Part 36. The characteristics of the aircraft with the engine size selected as described in the engine sizing paragraph are indicated on the figure. The performance of the baseline turbojet aircraft -5A, is also shown for reference.

Figure 3-25 presents the variation of operator's weight empty with engine size used for the mission performance calculations, the altitude for maximum range

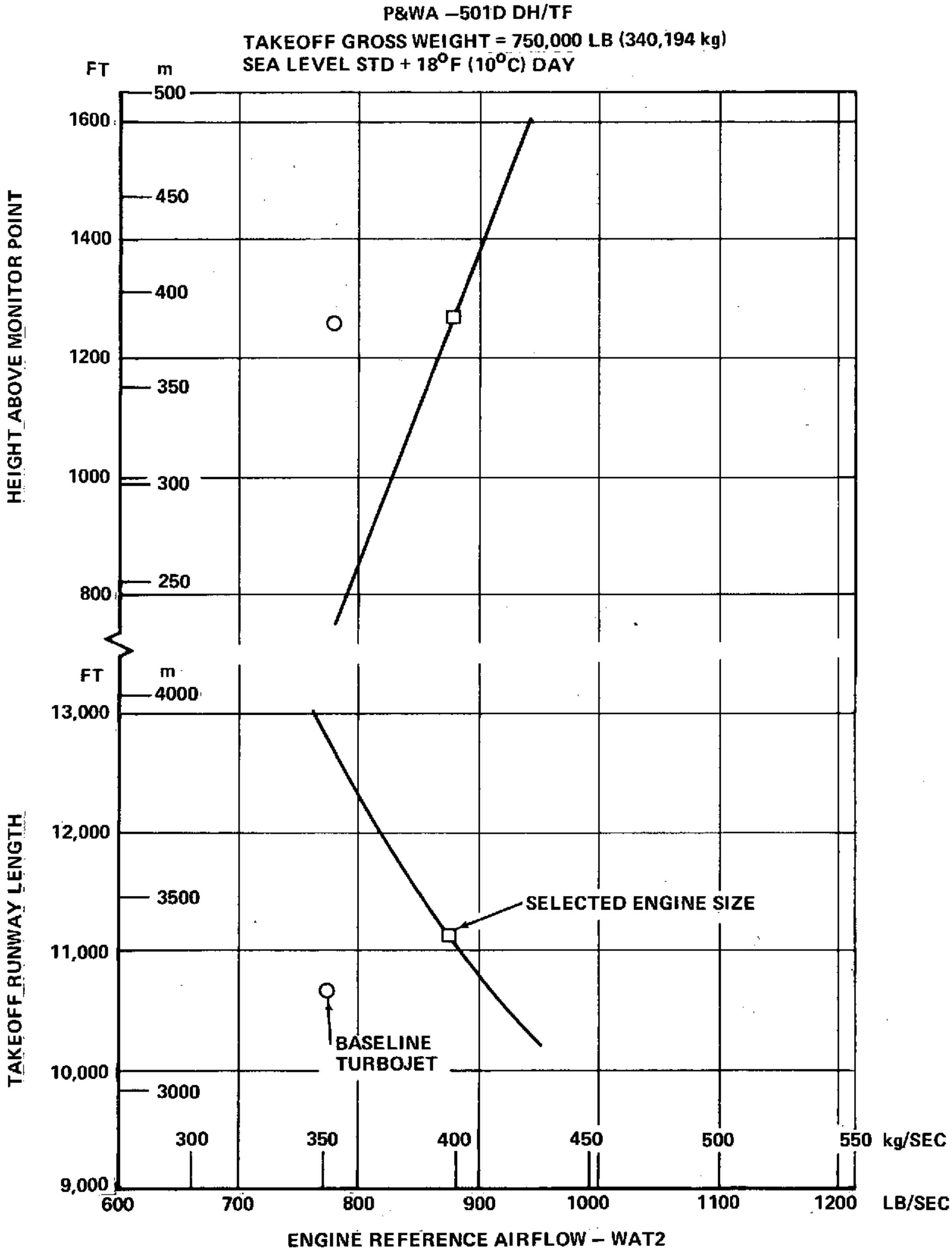


FIGURE 3-24. EFFECT OF ENGINE SIZE ON TAKEOFF PERFORMANCE

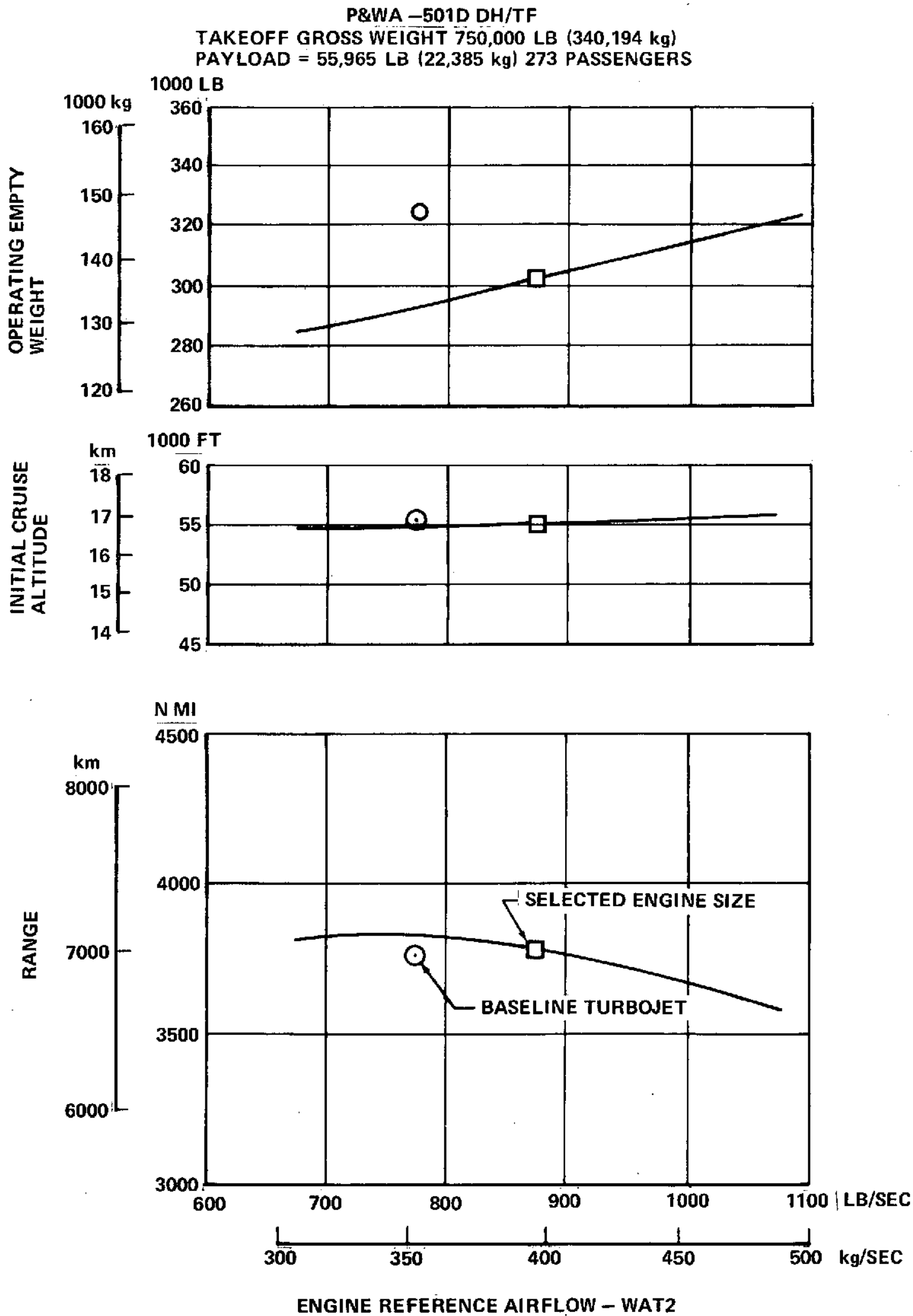


FIGURE 3-25. EFFECT OF ENGINE SIZE ON MISSION PERFORMANCE

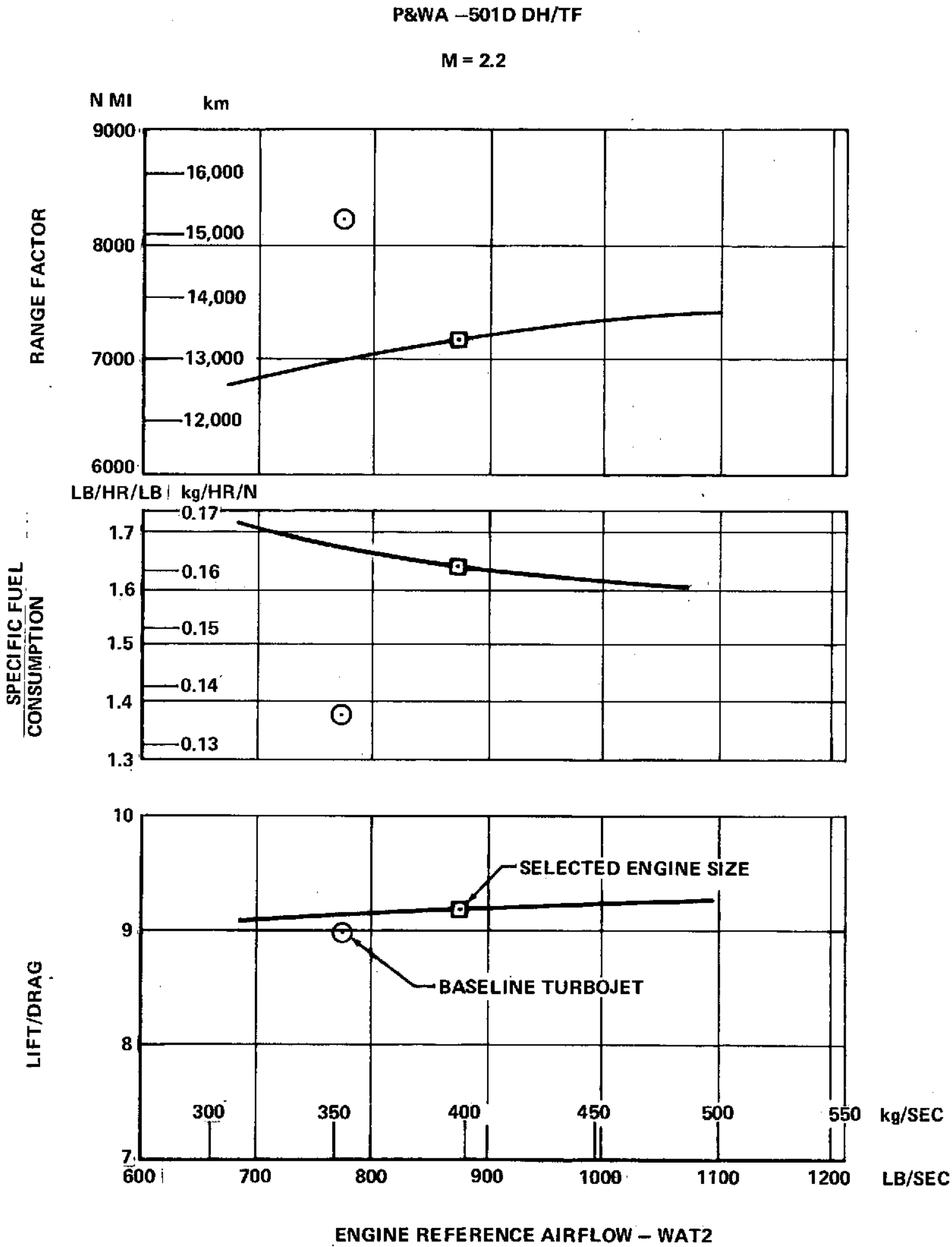


FIGURE 3-26. EFFECT OF ENGINE SIZE ON CRUISE PARAMETERS

factor at the start of the 2.2 M cruise, and the mission range. The selected engine size as indicated in the figure is identical to that specified in the engine sizing paragraph. A smaller size indicates better range, however, the engine cannot be reduced in size since the temperature of the fan stream impacting the suppressor is limiting. Figure 3-26 presents some of the details of the effect of engine size on the optimum cruise L/D, cruise installed SFC, and the 2.2 M cruise range factor.

The data presented in the last two figures accounts for the changes with engine size of engine and nacelle weight, and inlet and nacelle drags, but neglects the changes in aircraft wave drag. For a ten percent change in engine size, this effect is quite small, but can be significant for the larger engine sizes.

The performance for the -501D powered aircraft with the 875 lb/sec (397 kg/sec) engine is summarized below:

Takeoff Gross Weight	750,000 lbs. (340,194 kg)
Payload	55,965 lbs. (25,385 kg)
Takeoff Field Length	11,200 ft. (3,383 m)
Height at 3.5 n.mi. (6.5 km) Takeoff Point	1,268 ft. (386 m)
Range	3,790 n.mi. (7002 km)
Initial Cruise Altitude	55,258 ft. (16.8 km)
Direct Operating Cost (1973 $)	1.94 cents/seat n.mi.

The variation in range vs. initial subsonic leg length is shown in Figure 3-27. For a 600 n.mi. (1110 km) initial subsonic leg, the range penalty is 3 percent.

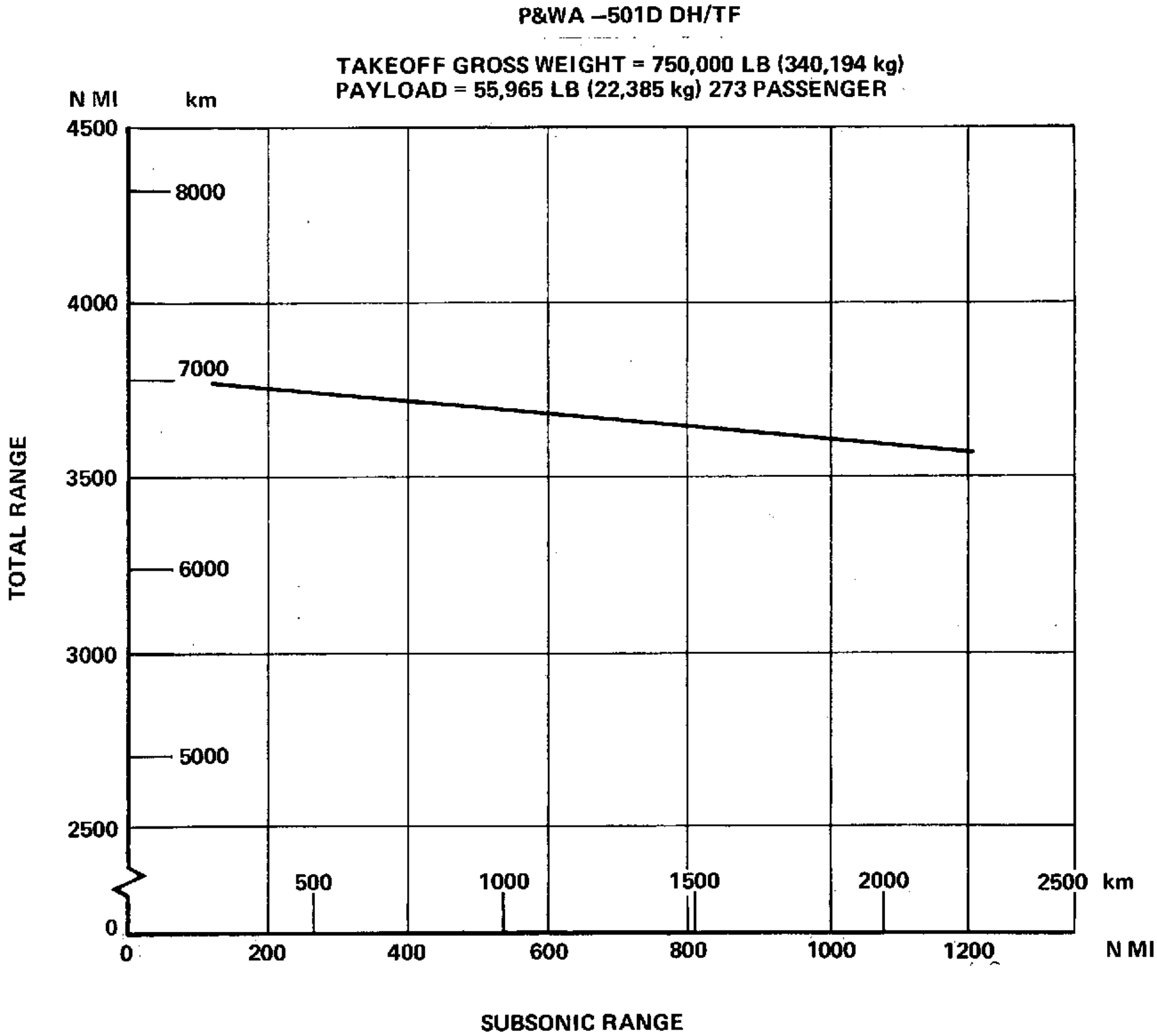


FIGURE 3-27. EFFECT OF INITIAL SUBSONIC LEG ON RANGE

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ENGINE SELECTION

The supersonic transport imposes unique requirements for an engine cycle. Ideally, this cycle would be a high bypass ratio turbofan with low jet velocity at takeoff and landing for low jet noise, a high bypass ratio turbofan at subsonic cruise and loiter for low SFC, a low bypass ratio turbofan or turbojet (augmentation optional) during climb/acceleration for high thrust and a mini-bypass turbojet during supersonic cruise for minimum SFC. To incorporate these diverse cycle requirements in a single package, a number of engine concepts, called variable cycle engines (VCE), are being defined by the engine manufacturers. This section includes analysis and airplane integration studies to evaluate the potential of currently identified VCE concepts.

Within the time period of this study, data for three candidate variable engines became available. These were all P&WA dual valve designs identified as the 201A, 201B and 302B. The 201A and 201B are three nozzle stream engines with the two duct streams augmented. The 302B is a two-nozzle stream unaugmented engine. During this period, GE did not provide a candidate variable cycle engine.

The three candidate engines presented unique problems in determining which to select for airplane integration. A sizing study and a preliminary airplane integration and performance exercise has been performed for each candidate to insure selection of the most promising for comparison with the mini-bypass and duct heating turbofan configurations. The following paragraphs present the results of these studies and information on the dual valve engine finally selected.

ENGINE SIZING

VCE 201A and 201B Sizing

The data for the augmented variable cycle engine are based on the P&WA Task VII engine, identified as VCE 201. This engine is a dual-valve, three-nozzle stream configuration with augmentation in the two fan streams. It is designed for Mach 2.2 to 2.7 supersonic cruise operation (configured by DAC in this application for Mach 2.2 supersonic cruise) and reflects P&WA technology levels consistent with initial operational capability in the late 1980's.

The engine incorporates a high bypass mode, utilized for takeoff and subsonic cruise, which significantly increases the engine airflow and results in an inlet that is either oversized for the low bypass mode at supersonic cruise or requires auxiliary inlets. Two versions of the VCE 201 engine (A and B) are defined by P&WA to evaluate the increase of engine airflow in the low mode of operation - a base flow, 201A, and a high flow, 201B. The airflow schedules for these engines are shown in Figure 4-1.

The VCE 201 engine consists of several unique components in addition to those that constitute a conventional duct heating turbofan engine. Illustrated in Figure 4-2, these unique components are:

1. Second fan with rotating flow splitter (fan 2)
2. Forward annulus inverting valve (fwd valve)
3. Rear annulus inverting valve (rear valve)
4. Second Primary Burner (burner 2)
5. Additional single stage low pressure turbine (LPT 2)
6. Second duct heater for third nozzle stream
7. Auxiliary nozzle for third nozzle stream

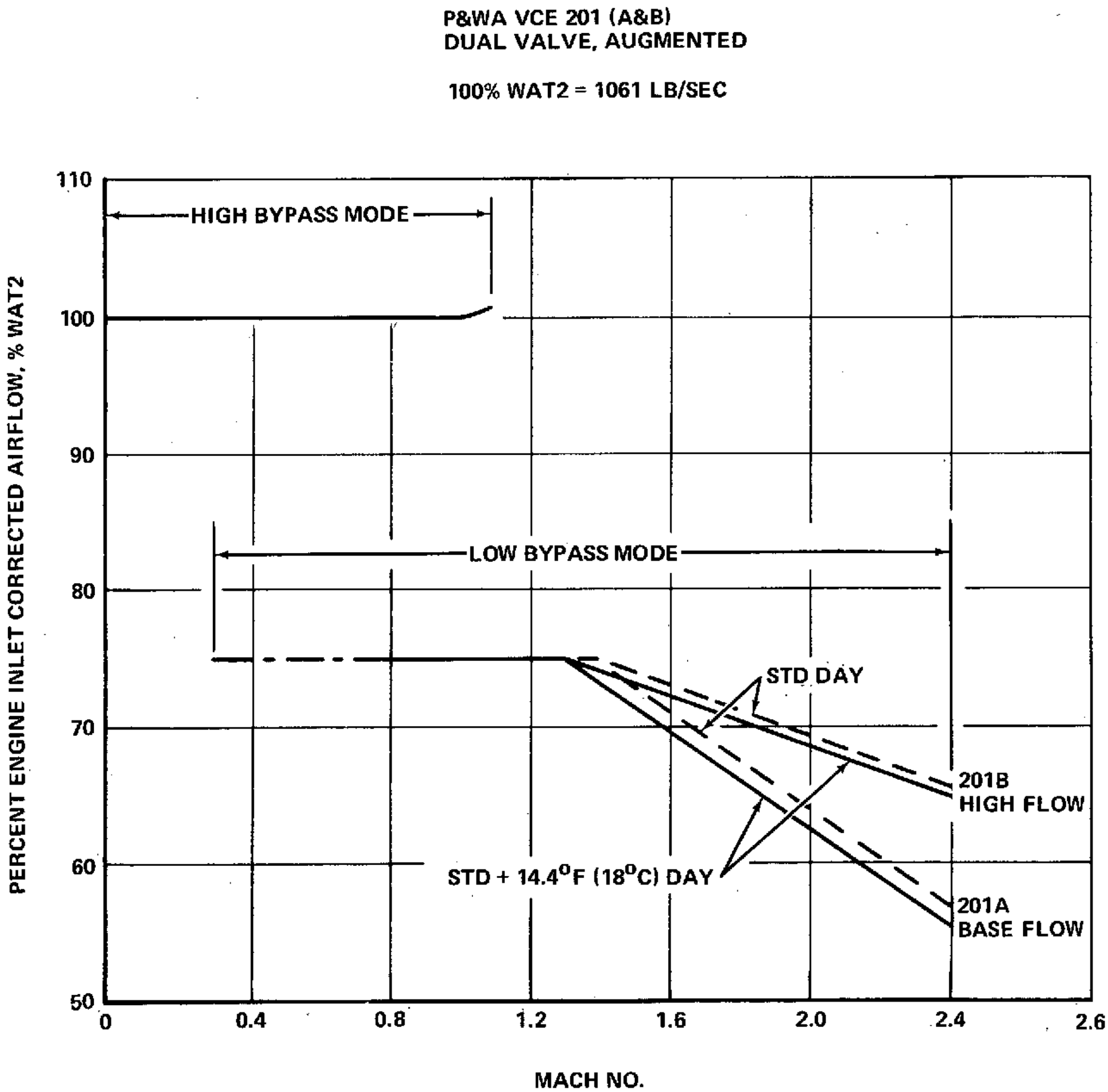

FIGURE 4-1. ENGINE INLET AIRFLOW SCHEDULE

AUGMENTED 3 NOZZLE STREAM
P&WA VCE-201

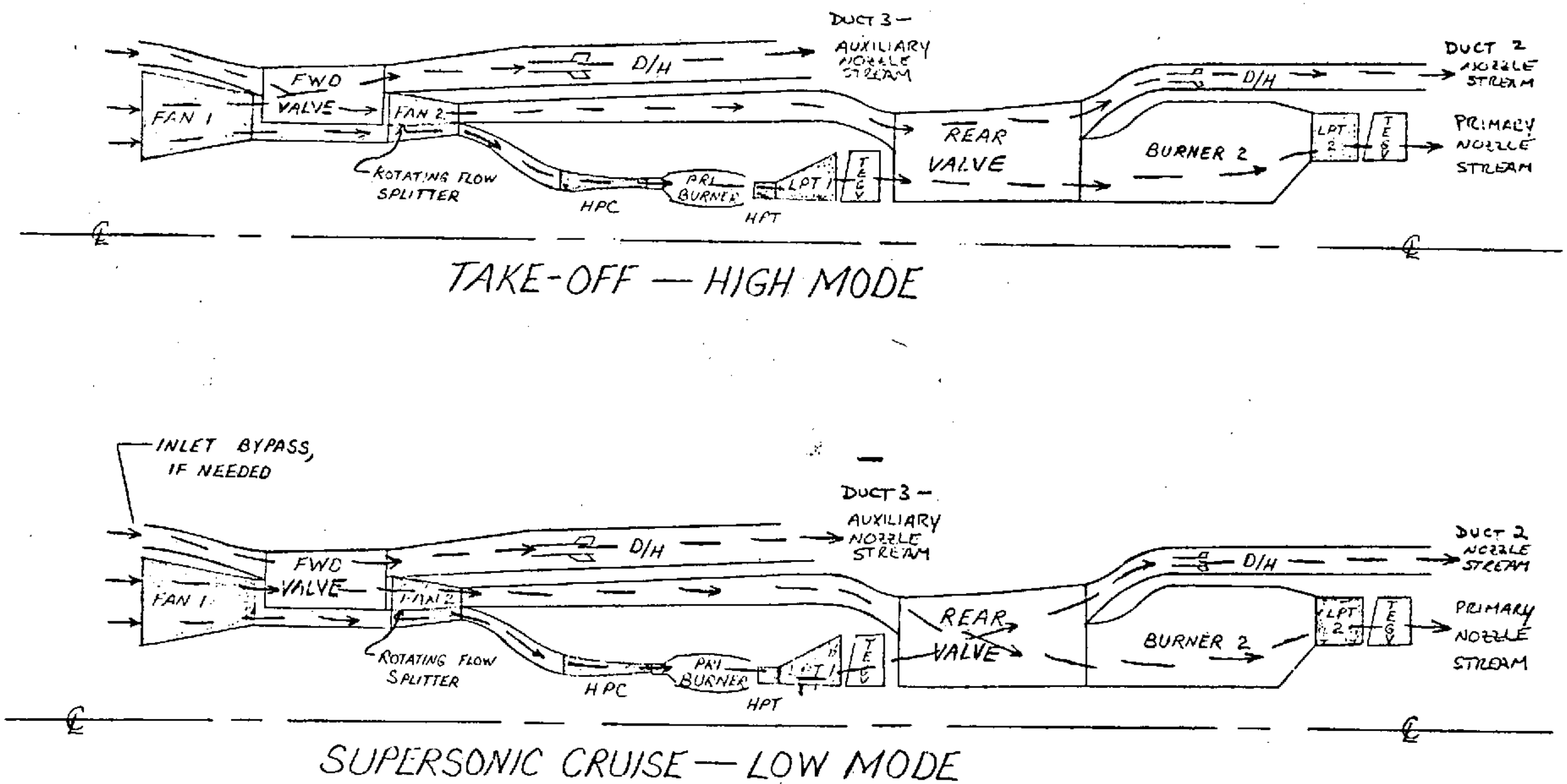

FIGURE 4-2. DUAL VALVE AUGMENTED VARIABLE CYCLE ENGINE SCHEMATIC

The engine is a twin-spool configuration with Fan 1, Fan 2, LPT 1 and LPT 2 all constituting the LP rotor spool. This arrangement allows the work to drive the two fans to be distributed between these two low pressure turbines in the most advantageous manner at each flight condition. The HP rotor spool consists of the HP compressor (HPC), primary combustor and the HP turbine (HPT). The engine has two duct stream augmentors as shown in Figure 4-2. The auxiliary nozzle stream is defined with the capability of converting from a convergent configuration for subsonic operation to a convergent-divergent configuration for supersonic operation in the high mode. In the low mode, the auxiliary nozzle is not utilized by the engine, therefore, in low mode operation during supersonic climb and cruise, the auxiliary nozzle can be used as an inlet bypass.

The 201A and 201B engines offer identical performance at takeoff. Therefore, this sizing study is applicable to both. Preliminary sizing criteria for the engines are takeoff thrust [52,000 lb/engine (231.3 kN), suppressed, uninstalled] FAR Part 36 noise for sideline [sea level, 0.3 Mach, 2100 ft. (640 m), sideline, Std. + 18°F (10°C) day] and for takeoff/cutback [33,250 lbs. (147.9 kN) thrust/ engine, uninstalled, 1050 ft. (320 m) 0.3 Mach, Std. + 18°F (10°C) day] and a suppressor temperature limit of 1500°F (1089°K). Figure 4-3 illustrates the engine sizing logic based on engine size, P&WA suppressor temperature limits, P&WA supplied single engine jet noise at the sideline (supplemented with DAC-generated four engine jet noise at the sideline) and the takeoff thrust requirement noted above. Data are shown for no suppressors and for finger type suppressors on both duct streams for various duct heat temperatures. P&WA suppressor loss data are utilized to determine the takeoff thrust required, see Figure 3-7.

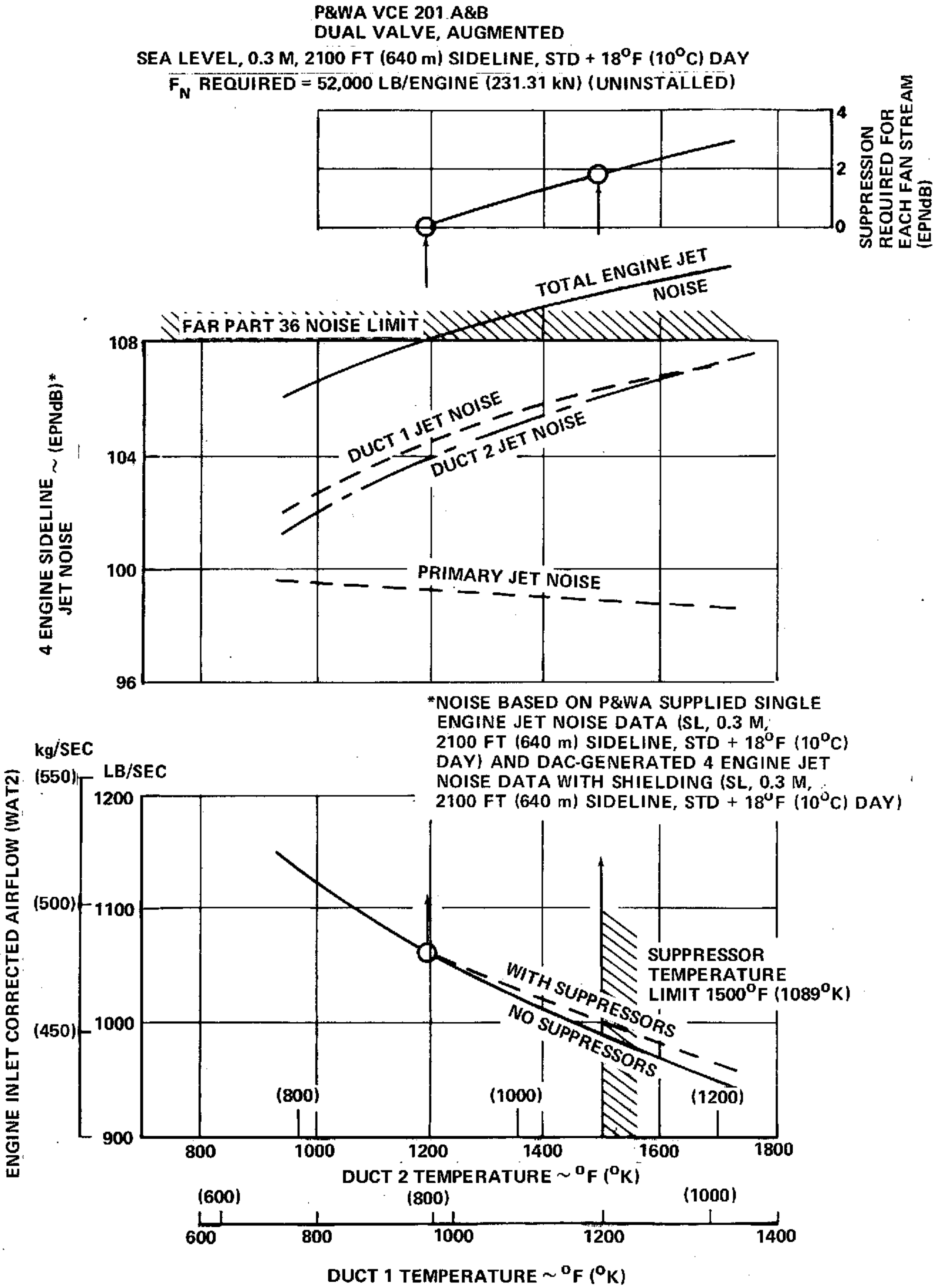

FIGURE 4-3. ENGINE SIZING FOR TAKEOFF

Examination of Figure 4-3 shows that the minimum sized engine to meet FAR Part 36 noise levels is a 1001 lb/sec (454.1 kg/sec) inlet corrected airflow engine providing 52,000 lbs. (231.3 kN) of thrust, suppressed, and employing 1.8 EPNdB suppressors on both duct streams. Note that the 1500°F (1089°K) suppressor temperature limit is critical for duct stream 2, duct stream 1 being 300°F (166°K) cooler and, therefore, not a constraint. However, the noise levels of the two duct streams are about equal, thereby requiring equivalent suppression. The question arises as to whether this engine, configured with two modest suppressors would indeed offer any advantage over an engine sized so as not to require suppressors. Again referring to Figure 4-3, it is shown that an engine sized to meet FAR Part 36 noise levels without suppressors is a 1061 lb/sec inlet corrected airflow engine.

Before proceeding with the engine size selection, noise at takeoff/cutback is examined. Figure 4-4 illustrates four engine takeoff/cutback noise (DAC estimated) at 1050 ft (320 m) altitude over the 3.5 n.mi. (6.5 km) monitor point for the two takeoff-sized engines, 1001 and 1061 lb/sec (454.1 and 481.3 kg/sec) inlet design corrected airflow. At the cutback thrust requirement of 33,250 lb/engine (147.9 kN) it is shown FAR Part 36 noise levels are not exceeded for either the 1001 lb/sec (454.1 kg/sec) or the 1061 lb/sec (481.3 kg/sec) sized engines with the unsuppressed noise being 107.1 EPNdB and 106.5 respectively. Therefore, noise at cutback is not a constraint for engine sizing.

In addressing the engine size recommended for initial studies, the question arises as to whether the smaller 1001 lb/sec (454.1 kg/sec) engine, employing jet noise suppressors on both duct streams, really has a physical size advantage over the 1061 lb/sec (481.3 kg/sec) engine which requires no suppressors at all. Table 4-1 shows the results of a trade study, comparing

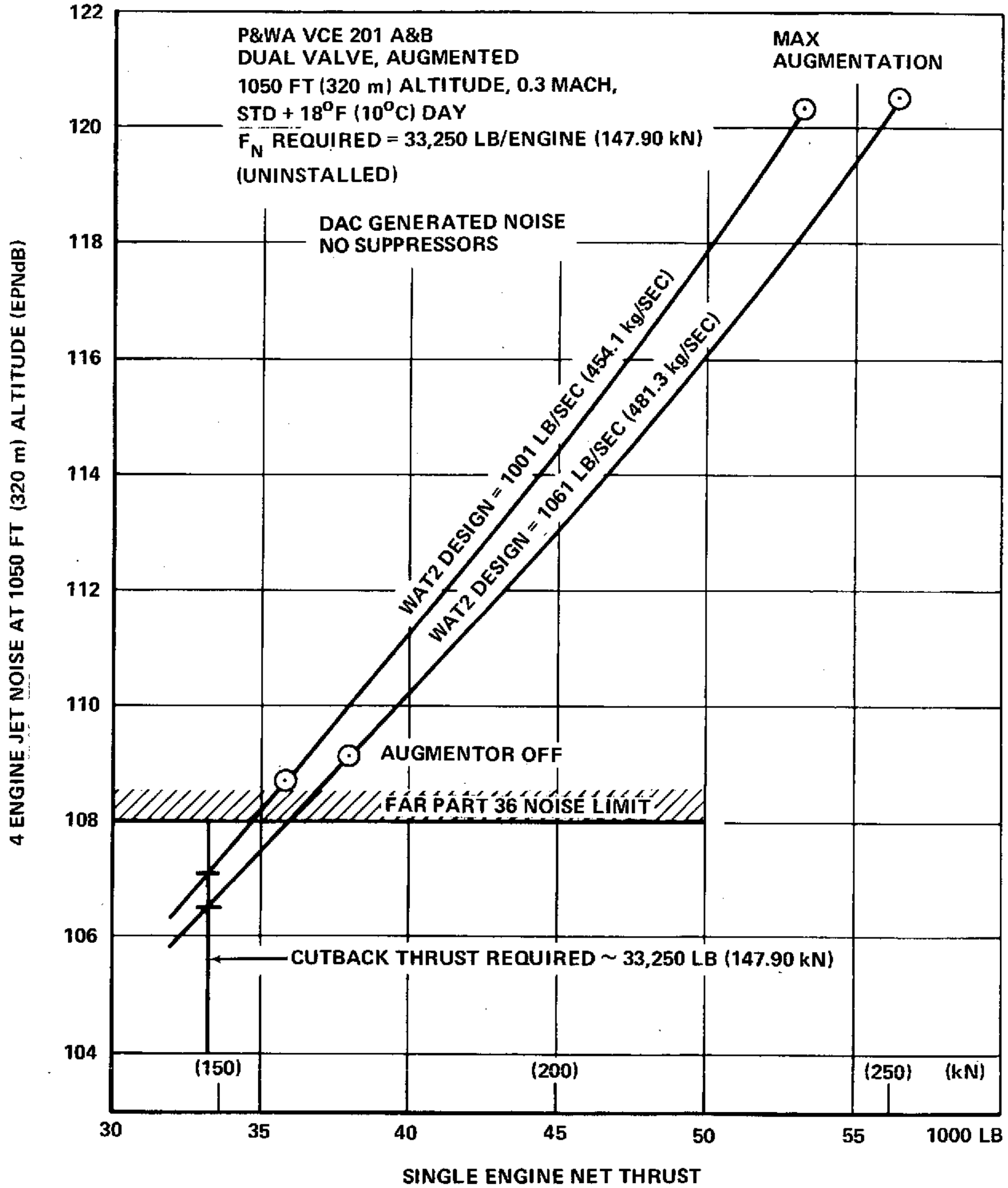

FIGURE 4-4. INFLIGHT NOISE CHARACTERISTICS

TABLE 4-1
COMPARISON OF P&WA VCE-201 ENGINE WITH AND WITHOUT SUPPRESSORS

	WEIGHT		INLET GAS FLOW DIA		MAXIMUM DIA		LENGTH	
	201A	201B	201A	201B	201A	201B	201A	201B
WITH SUPPRESSION 1001 PPS SIZE (454.1 kg/SEC)								
1500°F (1089°K)(FAN STREAM 2)								
107.5 PNdB – 2100 FT (640m) SIDELINE – SINGLE ENG	18592 LB	19936 LB	82.6 IN. (2.098 m)	79 IN. (2.007 m)	94.8 IN.	94.8 IN.	396 IN.	419 IN.
TWO FAN DUCT SUPPRESSORS	+1500 LB	+1500 LB			+4 IN.	+4 IN.	+40 IN.	+40 IN.
1.8 PNdB EACH TO MEET FAR 36	20092 LB (9113.7 kg)	21436 LB (9723.4 kg)			98.8 IN. (2.510m)	98.8 IN. (2.510m)	436 IN. (11.074m)	459 IN. (11.659m)
WITHOUT SUPPRESSION 1061 PPS SIZE (481.3 kg/SEC)								
1195°F (919°K) (FAN STREAM 2)								
105 PNdB – 2100 FT (640m) SIDELINE – SINGLE ENG	19854 LB (9005.8 kg)	21289 LB (9656.7 kg)	85.1 IN. (2.162m)	81.5 IN. (2.070m)	97.7 IN. (2.482m)	97.7 IN. (2.482m)	406 IN. (10.312m)	429 IN. (10.897m)
MEETS FAR 36								

the size and weight of the two sized engines (both A and B versions). Based on parametric data and information from P&WA, it is estimated that the two duct suppressors add 1500 lb. (680.4 kg) to the engine weight, 4 inches (10.2 cm) to the maximum diameter over the exhaust nozzle and 40 inches (101.6 cm) to the engine length. As a result, the 1061 lb/sec (481.3 kg/sec) engine without suppressors is approximately one percent lighter, one percent smaller in diameter, and one percent shorter in length. Consequently, the 1061 lb/sec (481.3 kg/sec) size engine with no suppression is selected for preliminary configuration development and initial mission studies.

P&WA indicates that a 10 percent higher-than-design airflow feature at takeoff could be used for the 201A and 201B engines; however, sufficient data were not available for including this feature in the study.

The nozzles for the 201 engine are variable area type (variable throat and exit areas) containing an integral thrust reverser and ejector on the primary and one fan stream. No thrust reverser or ejector is included for the auxiliary nozzle stream. No jet noise suppressor is incorporated. The primary and both fan duct throat areas are variable. The engines including the P&WA nozzles are shown in Figures 4-5 and 4-6, and the installed engines, including auxiliary inlets, are shown in Figures 4-7 and 4-8. To accommodate the high flow at supersonic cruise of the 201B, the 201A hub-to-tip diameter ratio is reduced from .381 to .33 to increase the inlet annulus flow area. However, as this causes a reduction in overall pressure rise across the fan, a third fan stage is added to both 201B fans to maintain overall design fan pressure ratio of 2.5.

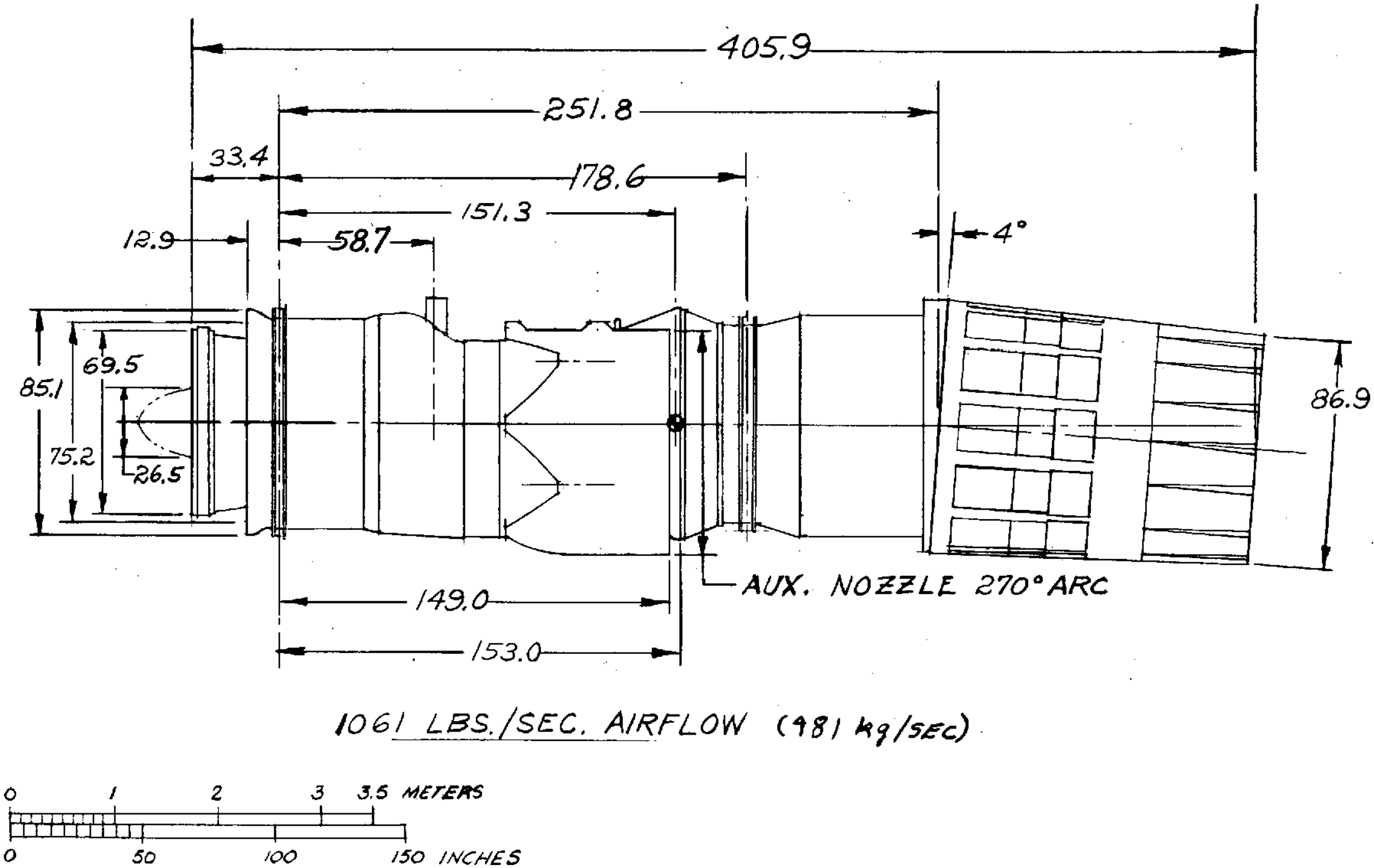

FIGURE 4-5. P&WA 201A VARIABLE CYCLE ENGINE

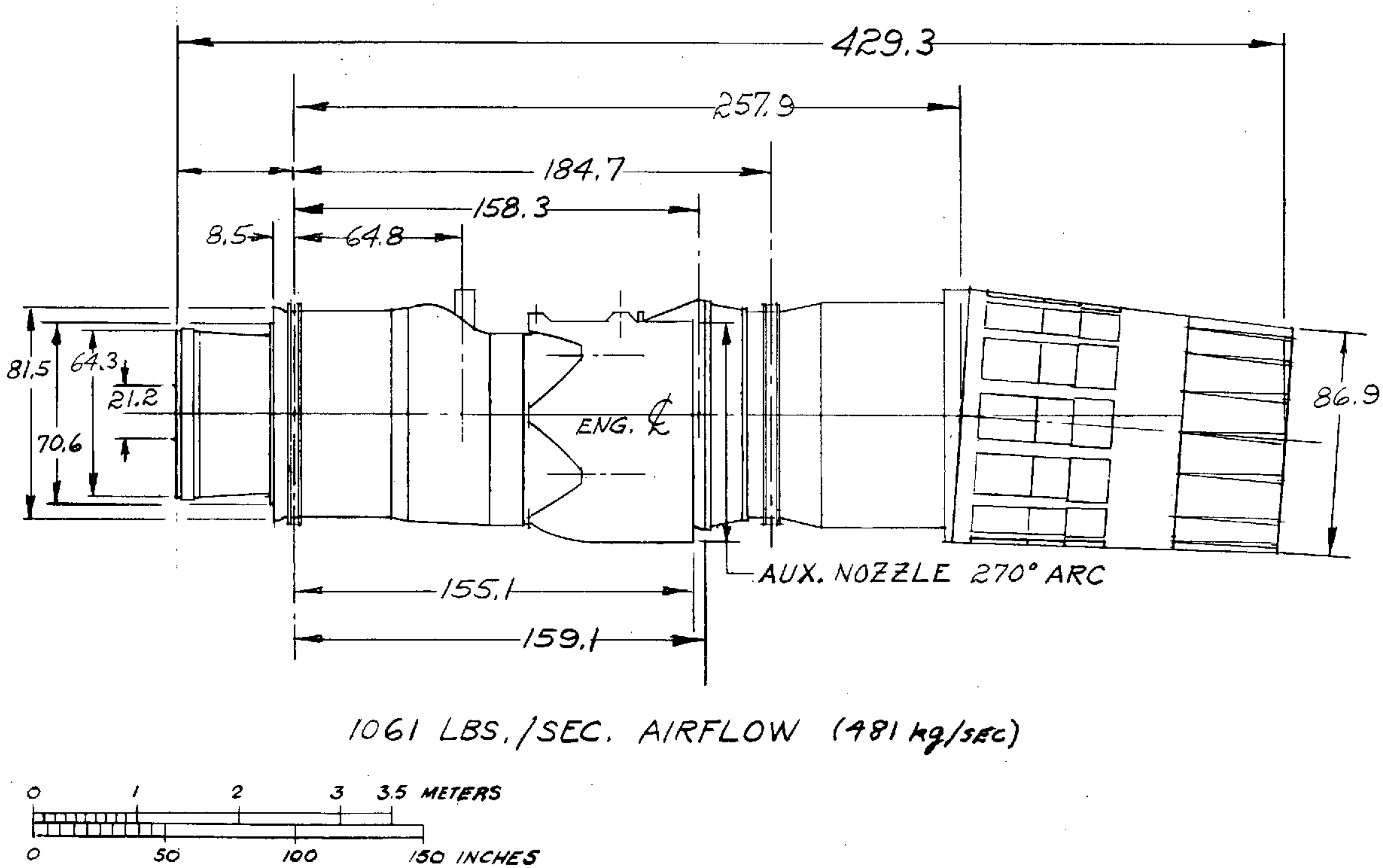

FIGURE 4-6. P&WA 201B VARIABLE CYCLE ENGINE

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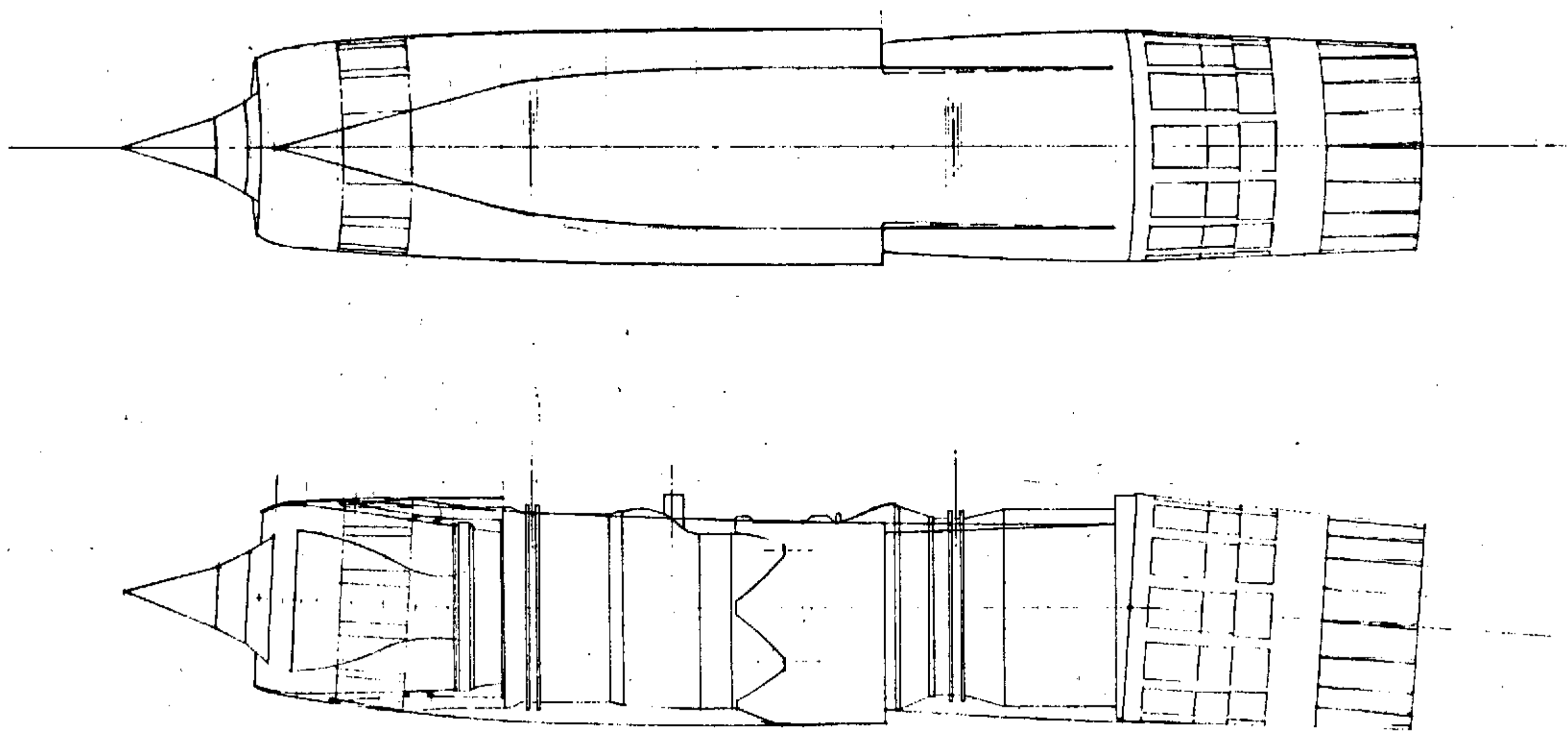


1061 LB/SEC AIRFLOW (481 kg/SEC)

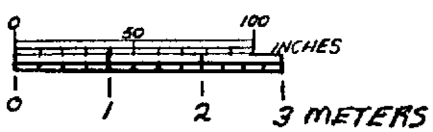
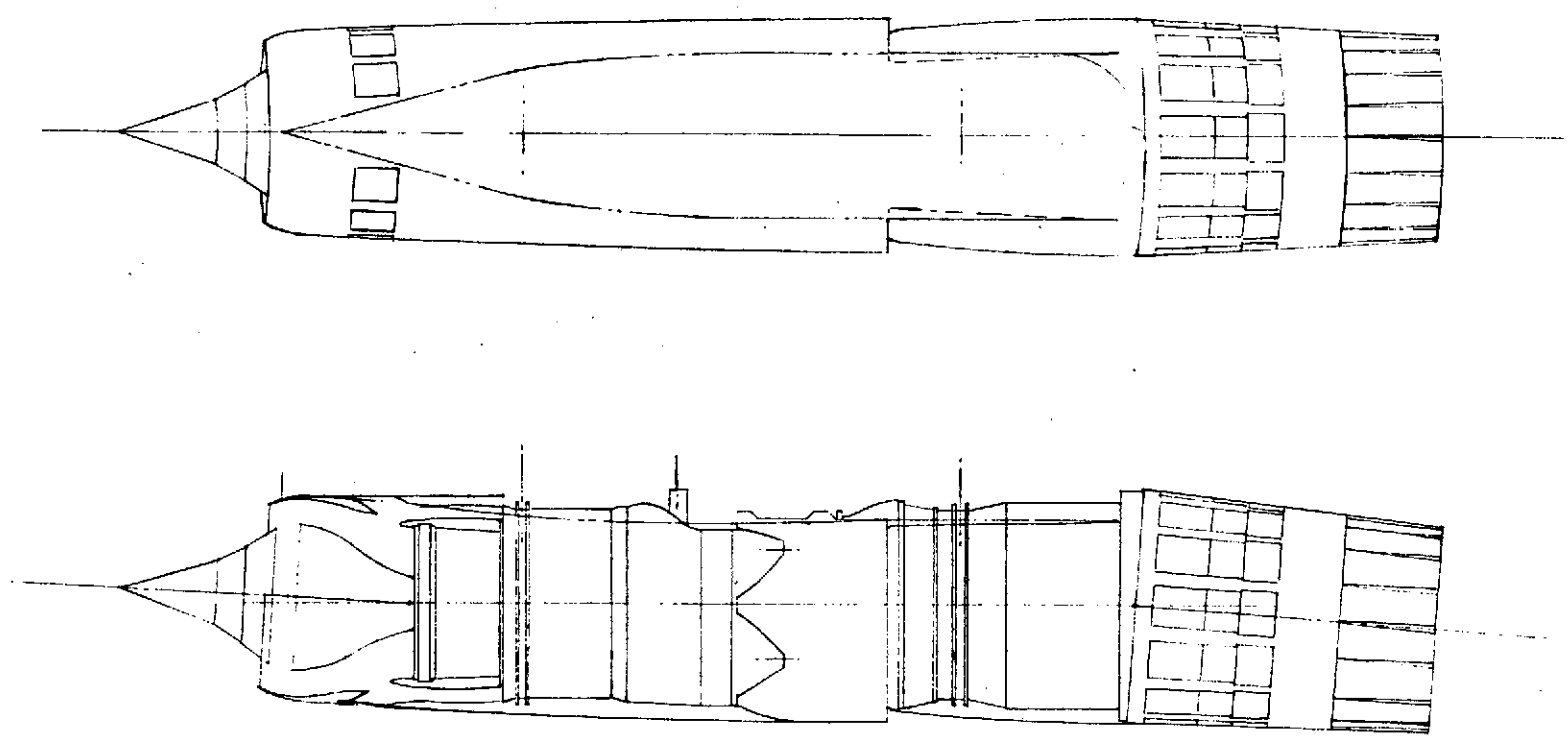


FIGURE 4-7. P&WA 201A VARIABLE CYCLE ENGINE INSTALLATION



1061 LB/SEC AIRFLOW (481 kg/SEC)

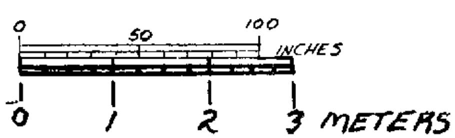


FIGURE 4-8. P&WA 201B VARIABLE CYCLE ENGINE INSTALLATION

Engine weights, dimensions, scaling equations and cost data are presented in Table 4-2. The cost data are based on P&WA cost information provided as part of the Advanced Supersonic Propulsion System Technology Studies conducted under contract to NASA Lewis in 1973. Costs have been escalated to 1973 by DAC based on 1972 dollar values provided by the engine manufacturers' study.

VCE 302B Sizing

The data for the unaugmented variable cycle engine are based on the P&WA Task VII engine identified as 302B. This engine is a dual-valve, two-nozzle stream configuration with no duct heating augmentation. It is designed for Mach 2.2 to 2.7 supersonic cruise operation (configured by DAC in this application for Mach 2.2 supersonic cruise) and reflects technology levels consistent with initial operational capability in the late 1980's.

The 302B engine is identical to the 201 engine except that the duct burners and the third nozzle stream are deleted as illustrated in Figure 4-9. The engine has a twin-spool configuration with Fan 1, Fan 2, LPT and LPT 2 all constituting the LP rotor spool. This arrangement allows the work to drive the two fans to be distributed between these two low pressure turbines in the most advantageous manner for each flight condition. The HP rotor spool consists of the HP compressor (HPC), primary combustor, and the HP turbine (HPT).

For takeoff operation, the forward valve is in the cross-over position (top sketch in Figure 4-9). In this mode, air flows from the first fan to the outer duct and inlet air crosses directly into the second fan. The rear valve is also in the cross-over position. This mode yields minimum noise levels while providing maximum thrust by allowing the second burner to serve as a thrust augmentor in place of duct-heaters. For supersonic operation, the first valve is in the straight-through position and the second valve is again

TABLE 4-2

P&WA VCE 201A AND B ENGINE CHARACTERISTICS SUMMARY
1061 LB/SEC (481 kg/SEC) RATED AIRFLOW

DESIGN CYCLE CHARACTERISTICS

BYPASS RATIO	3.0
FAN PRESSURE RATIO	
FAN 1	2.5
FAN 2	2.5
CYCLE PRESSURE RATIO	20
COMBUSTOR EXIT TEMP	2600°F (1700°K)

TAKEOFF RATINGS [STD DAY + 18°F (10°C)]

MAX THRUST (SLS) – LB (kN)	60,375 (268.56)
MAX THRUST (SL, 0.3 M, UNINSTALLED) – LB (kN)	55,240 (245.72)

WEIGHT (LB)	201A	201B
ENGINE – LB (kg)	15,787 (7161.0)	16,983 (7703.5)
ENGINE/NOZZLE/ REVERSER – LB (kg)	19,854 (9005.8)	21,289 (9656.7)

DIMENSIONS	201A	201B
ENGINE INLET GAS FLOW PATH DIA – IN. (m)	69.5 (1.765)	64.3 (1.633)
HUB-TO-TIP RATIO (AT PLANE OF ATTACH FLANGE)	0.381	0.33
ENGINE MAX DIA – IN. (m)	97.7 (2.482)	97.7 (2.482)
LENGTH – INLET FLANGE TO EXHAUST PLANE – IN. (m)	405.9 (10.310)	429.3 (10.904)

SCALING FACTORS

WEIGHT $$\frac{WT}{WT\ BASE} = \left(\frac{WAT2}{1061}\right)^{1.086}$$

DIAMETER $$\frac{D}{D\ BASE} = \left(\frac{WAT2}{1061}\right)^{0.50}$$

LENGTH $$\frac{L}{L\ BASE} = \left(\frac{WAT2}{1061}\right)^{0.39}$$

COST*

ENGINE/NOZZLE $5.63M

SCALING FACTOR $$\frac{COST}{COST\ BASE} = \left(\frac{WAT2}{1061}\right)^{0.53}$$

*BASED ON:
- 1973 DOLLARS
- 1980 ENGINE TECHNOLOGY
- PRICES INCLUDE ALL DEVELOPMENT COSTS PLUS FIVE-YEAR PRODUCT SUPPORT AFTER CERTIFICATION, BASED ON ONE-ENGINE MODEL
- 3000-ENGINE PRODUCTION RUN

NONAUGMENTED 2 NOZZLE STREAM
P&WA VCE-302B

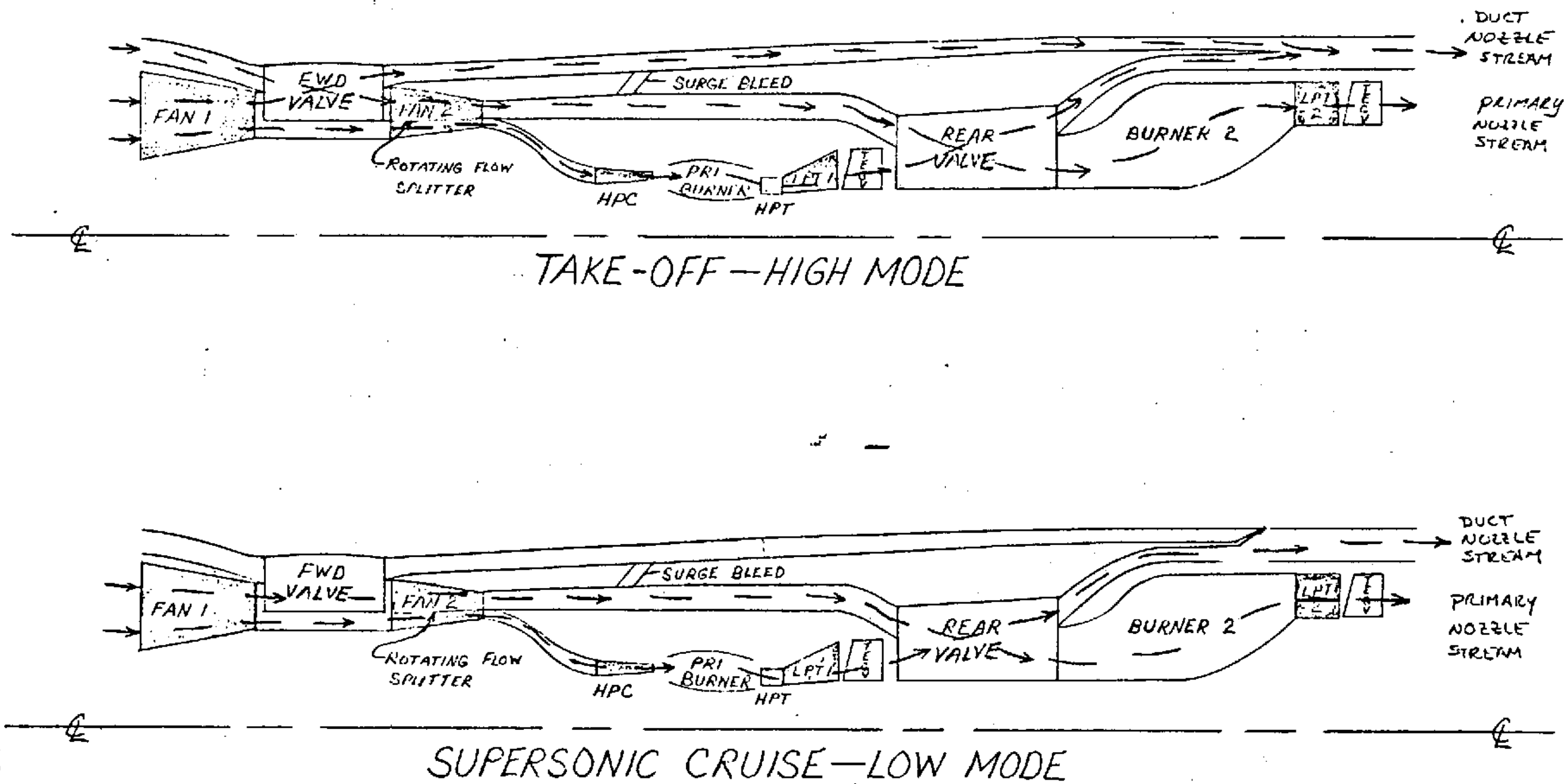


FIGURE 4-9. DUAL VALVE NONAUGMENTED VARIABLE CYCLE ENGINE SCHEMATIC

crossed. This converts the engine to two turbojet cycles as shown in the bottom sketch of Figure 4-9. For subsonic cruise and loiter operation, the first valve is crossed and the second is straight-through. For this mode, the second burner is off. This combination minimizes the fuel consumption level for these intermediate and low thrust operating conditions.

Other features of this engine include:

- High engine airflow at supersonic cruise which allows the engine to better match airflow characteristics of the inlet between takeoff and supersonic cruise.
- Ten percent higher-than-design airflow at static to 0.3 Mach to decrease jet noise for takeoff.

Preliminary sizing criteria for this engine are the same as for the 201. Figure 4-10 illustrates the engine sizing logic showing the resultant engine size airflow and P&WA supplied single engine jet noise at 2100 ft (640 m) sideline [modified by DAC to simulate four engine jet noise at this condition, sea level, 0.3 Mach, Std. + 18°F (10°C) day] for varying percent of takeoff thrust at a takeoff thrust requirement of 52,000 lb/engine (231.3 kN). Temperature of the duct exhaust stream, being non-augmented, does not exceed 745°F (670°K), thereby not imposing a constraint for suppressors, should they be required.

Examination of Figure 4-10 shows that FAR Part 36 sideline noise level is not exceeded, even at full throttle. Maximum four engine jet noise at sideline is 107.9 EPNdB at the 100% takeoff power setting. At the optional 10 percent higher-than-design airflow power setting offered by P&WA (note Figure 4-11), the noise at sideline is 106.9 EPNdB. Therefore, at sideline, no suppressors are required and the engines can be sized at full throttle with no cutback.

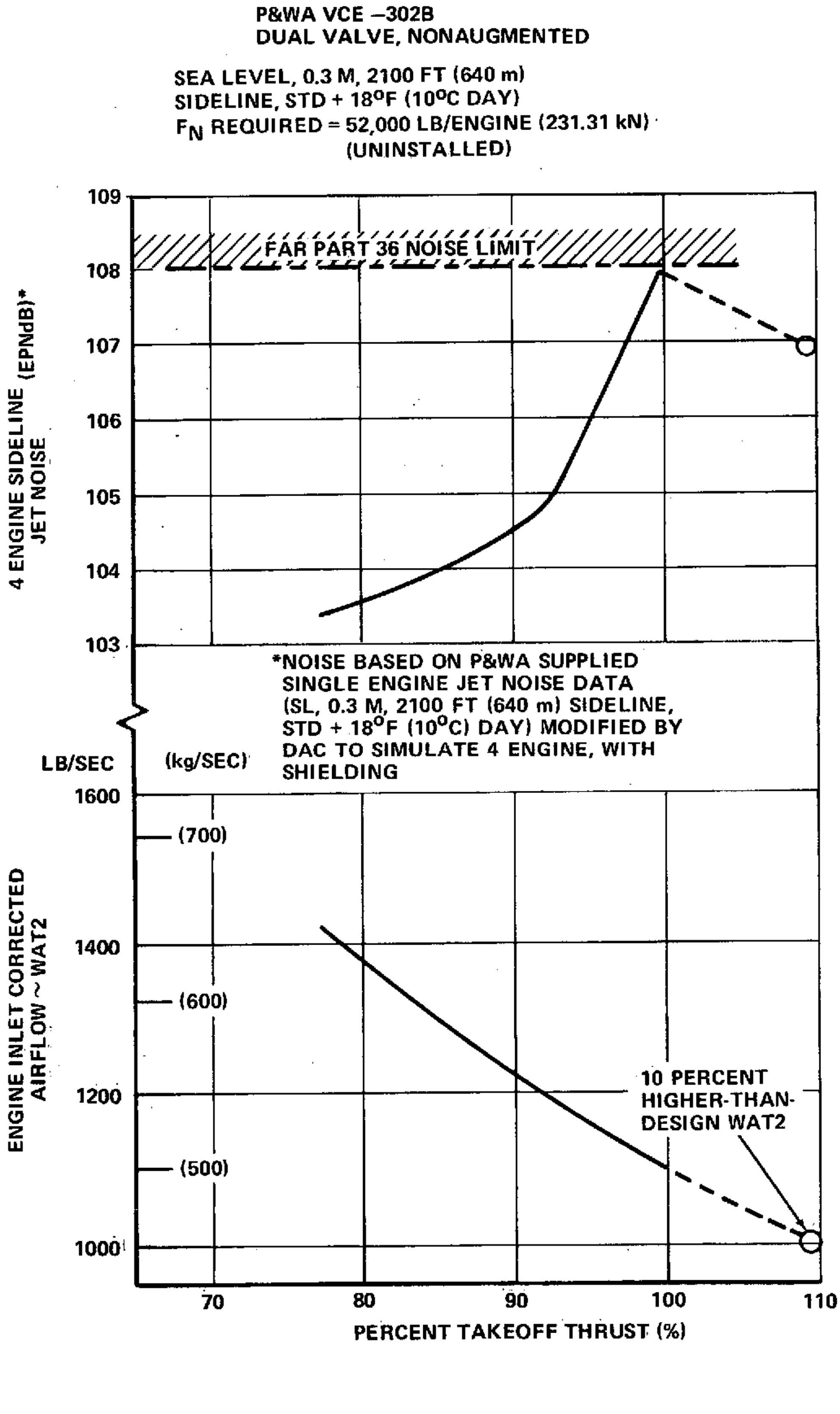


FIGURE 4-10. ENGINE SIZING FOR TAKEOFF

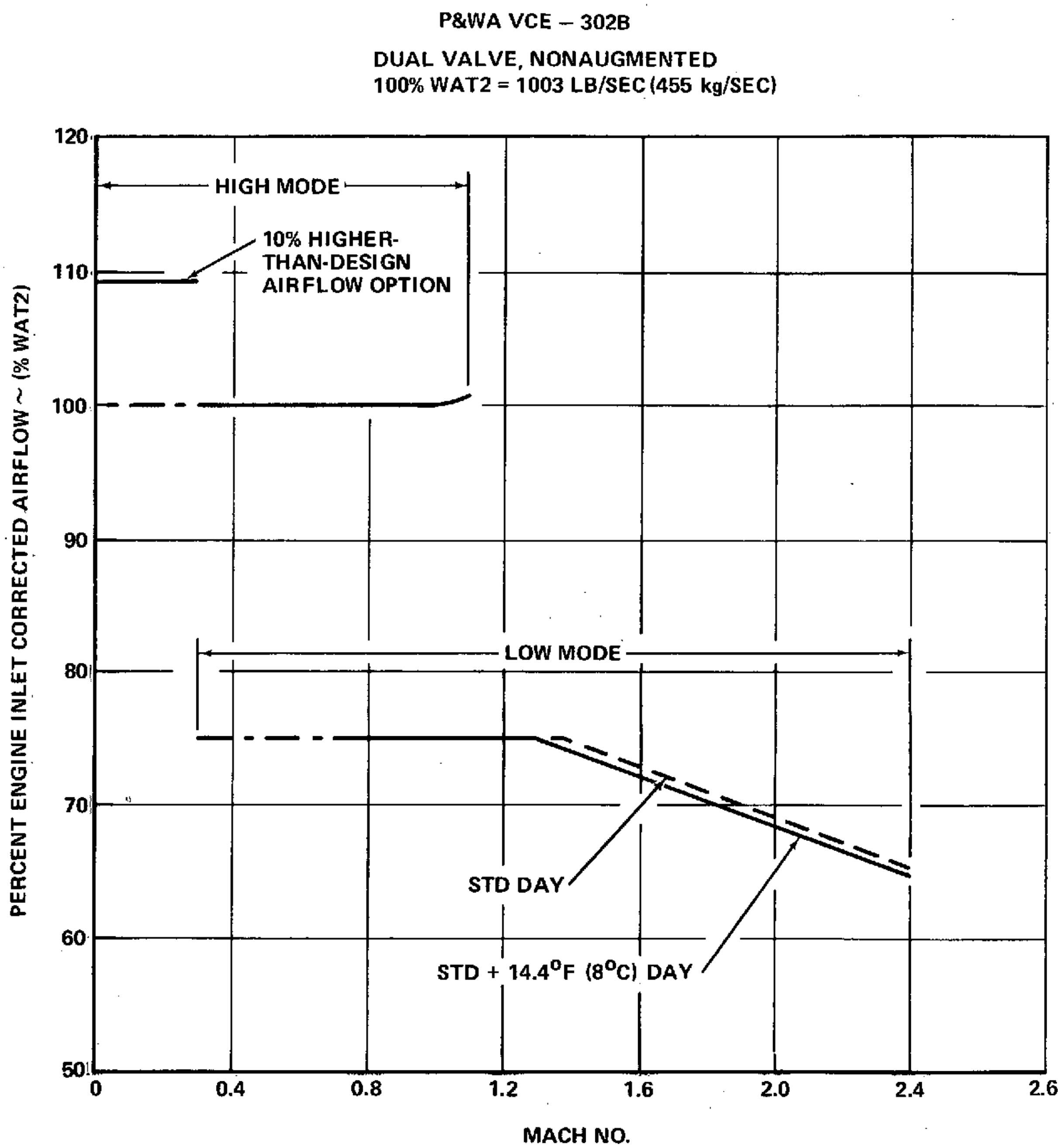


FIGURE 4-11. ENGINE INLET AIRFLOW SCHEDULE

At the 100% takeoff lb/sec power setting, the engine size required for 52,000 lb. (231.3 kN) takeoff thrust is 1097 lb/sec (497.6 kg/sec) design corrected airflow. Utilizing the 10 percent higher-than-design airflow feature, the engine size required is reduced to 1003 lb/sec (455 kg/sec) design corrected airflow.

Noise at takeoff/cutback has been examined. Figure 4-12 illustrates four engine takeoff/cutback noise (DAC estimated) at 1050 ft. (320 m) altitude over the 3.5 n.m. (6.5 km) monitor point for the two takeoff-sized engines, 1097 and 1003 lb/sec (497.6 and 455 kg/sec) inlet design corrected airflow. At the cutback thrust requirement of 33,250 lb/engine (147.9 kN), it is shown that FAR Part 36 takeoff noise level is not exceeded, four engine jet noise for the 1097 lb/sec (497.6 kg/sec) and the 1003 lb/sec (455 kg/sec) sized engines are 105.4 and 106.1 EPNdB respectively.

Summarizing, noise levels at sideline and takeoff/cutback are below the FAR Part 36 noise levels. Therefore, no suppressors are required and noise is not a constraint for engine sizing. The engine size of 1003 lb/sec (455 kg/sec) is used for the preliminary configuration development and initial mission studies. The minimum size engine is selected to conform with the prerequisite of working with engine manufacturers, using their data, and exploiting their estimated maximum technology projections.

The nozzles for this engine are variable area type (variable throat and exit area) containing an integral thrust reverser and ejector. Both the primary and fan duct throat areas are variable. No jet noise suppressor is incorporated. The engine including the P&WA nozzle is shown in Figure 4-13 and the installed engine, including auxiliary inlet, is shown in Figure 4-14.

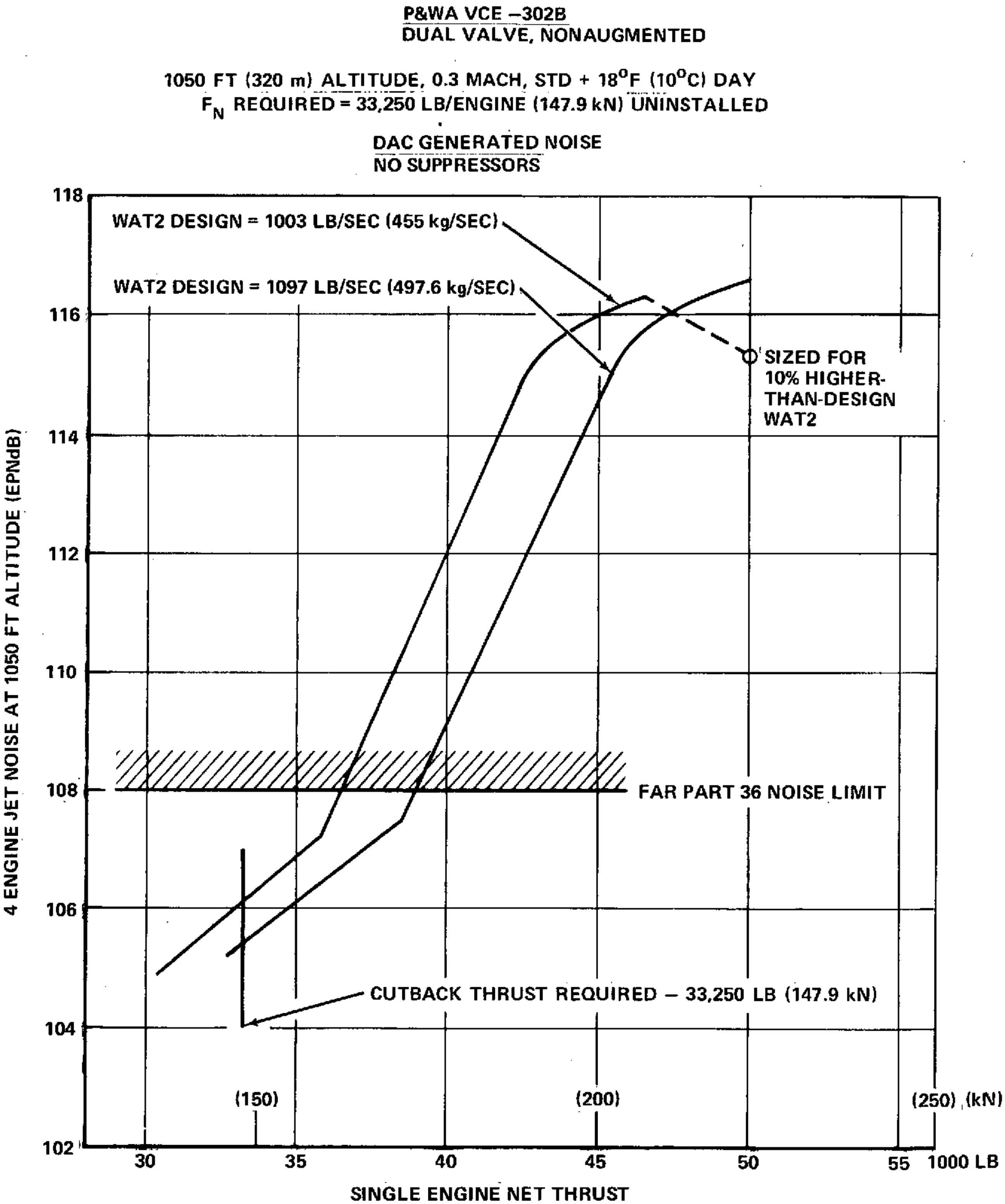
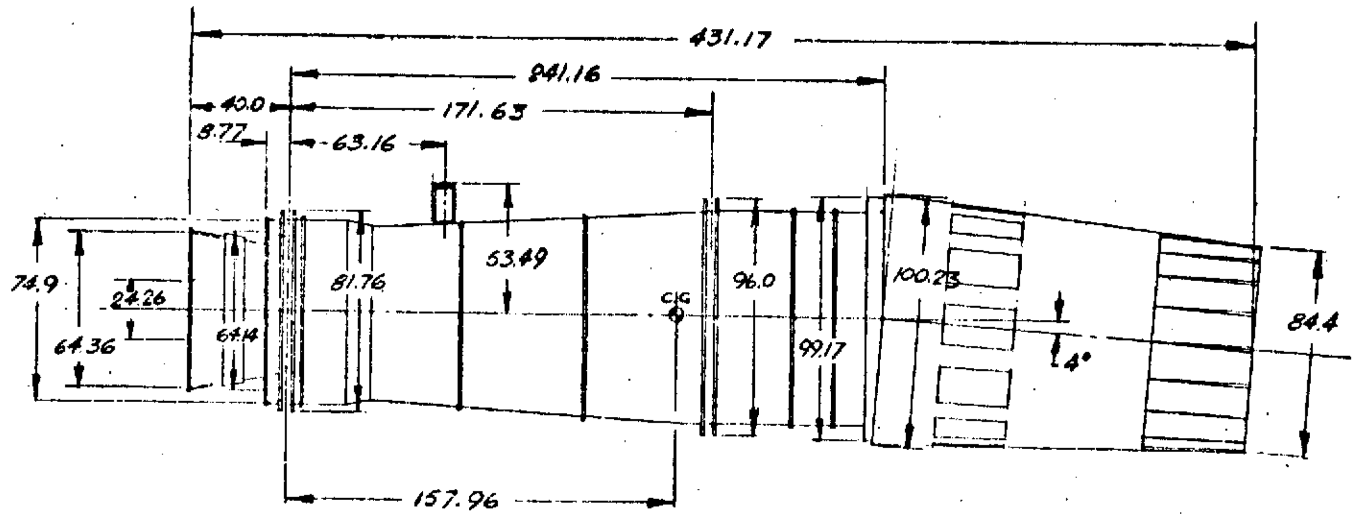


FIGURE 4-12. INFLIGHT NOISE CHARACTERISTICS



WT. = 19,575 LB.
AIRFLOW = 1003 LB/SEC (455 kg/SEC)

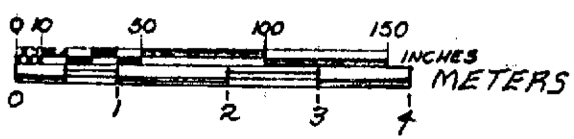


FIGURE 4-13. P&WA 302B VARIABLE CYCLE ENGINE

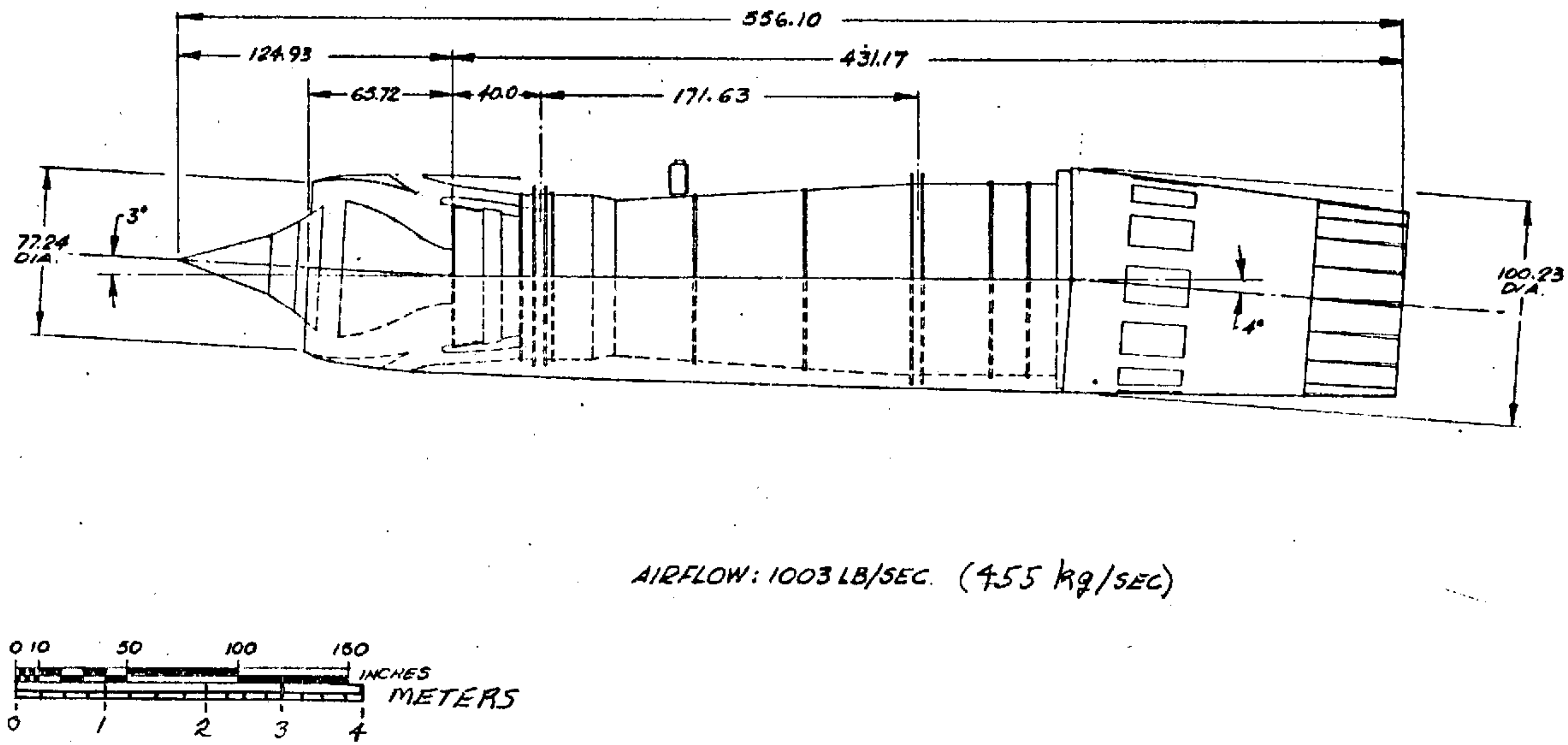


FIGURE 4-14. P&WA 302B VARIABLE CYCLE ENGINE INSTALLATION

Engine weights, dimensions, scaling factors, equations and cost data are presented in Table 4-3.

The cost data are based on P&WA cost information provided as part of their Advanced Supersonic Propulsion System Technology Studies conducted under contract to NASA Lewis in 1973. Costs have been escalated to 1973 by DAC based on 1972 dollar values provided by the engine manufacturers' study.

Final Engine Selection

A simplified performance evaluation has been conducted on the three engines described to select one engine for a more in-depth study required to compare the VCE with the other engines. The installed engine performance characteristics of each candidate engine have been generated with emphasis on installation effects which are peculiar to each engine. For example, the 201A and 201B engines each discharge the third stream during the high flow mode through a separate auxiliary nozzle wrapped around the nacelle. Since there is no flow through this nozzle during supersonic cruise, and there appeared to be no way to fair over the nozzle when it was not being used, an estimated base drag installation penalty of 3.83 drag counts (ΔC_D = .000383) is included for these engines.

For the purpose of this initial screening, it has been assumed that the wave drag and the structural weight penalty would be essentially the same for each engine when installed on the airplane. The mission performance with each engine has been determined, based on these assumptions, and using the estimated engine and nacelle weights. Table 4-4 compares the mission performance of the three VCE engines, based on the above assumptions, plus the estimated engine and nacelle weights. In addition, results of a trade study evaluating impact of augmentation during climb (for the augmented engines 201A and 201B) on mission performance are presented.

TABLE 4-3

P&WA VCE 302B ENGINE CHARACTERISTICS SUMMARY
1003 LB/SEC (455 kg/SEC) RATED AIRFLOW

DESIGN CYCLE CHARACTERISTICS

BYPASS RATIO (TAKEOFF/SUPERSONIC)	3.0/0
FAN PRESSURE RATIO	
FAN 1	2.5
FAN 2	3.0
CYCLE PRESSURE RATIO	20

COMBUSTOR EXIT TEMP ^oF (^oK) (1ST BURNER/2ND BURNER)	
TAKEOFF	2600/2600 (1700/1700)
MAX TRANSONIC CLIMB	2500/2500 (1644/1644)
MAX SUPERSONIC CRUISE	2400/2100 (1589/1422)

TAKEOFF RATINGS [STD DAY + 18^oF (10^oC)]

MAX THRUST (SLS) – LB (kN)	58,050 (258.22)
MAX THRUST (SL, 0.3 M, UNINSTALLED – LB (kN)	52,000 (231.31)

WEIGHT

ENGINE – LB (kg)	15,863 (7195.5)
ENGINE + NOZZLE/REVERSER – LB (kg)	19,575 (8879.2)

DIMENSIONS

ENGINE INLET GAS	
FLOW PATH DIA – IN. (m)	64.36 (1.635)
HUB-TO-TIP RATIO (AT PLANE OF ATTACH FLANGE)	0.377
ENGINE MAX DIA – IN. (m)	99.17 (2.519)
LENGTH – INLET FLANGE TO EXHAUST PLANE – IN. (m)	431.17 (10.952)

SCALING FACTORS

WEIGHT $$\frac{WT}{WT\ BASE} = \left(\frac{WAT2}{1003}\right)^{1.086}$$

DIAMETER $$\frac{D}{D\ BASE} = \left(\frac{WAT2}{1003}\right)^{0.50}$$

LENGTH $$\frac{L}{L\ BASE} = \left(\frac{WAT2}{1003}\right)^{0.39}$$

COST*

ENGINE/NOZZLE $4.89M

SCALING FACTOR $$\frac{COST}{COST\ BASE} = \left(\frac{WAT2}{1003}\right)^{0.53}$$

*BASED ON:
- 1973 DOLLARS
- 1980 ENGINE TECHNOLOGY
- PRICES INCLUDE ALL DEVELOPMENT COSTS, PLUS FIVE-YEAR PRODUCT SUPPORT AFTER CERTIFICATION, BASED ON ONE-ENGINE MODEL
- 3000-ENGINE PRODUCTION RUN

TABLE 4-4

ENGINE PERFORMANCE COMPARISON
TAKEOFF GROSS WEIGHT = 750,000 LB (340,194 kg)
PAYLOAD = 55,965 LB (25,385 kg)

ENGINE	201A			201B			302B
ENGINE SIZE, LB/SEC	1061 (481 kg/SEC)			1061 (481 kg/SEC)			1003 (455 kg/SEC)
OPERATING EMPTY WEIGHT, LB	338,620 (153,595 kg)			344,356 (156,197 kg)			345,867 (156,883 kg)
CLIMB THRUST	NO AUG	MIN AUG	PARTIAL AUG	NO AUG	MIN AUG	PARTIAL AUG	MAX CLIMB
RANGE, N MI	2983 (5525 km)	2982 (5523 km)	2973 (5506 km)	2848 (5274 km)	2857 (5291 km)	2845 (5269 km)	3088 (5719 km)

Of the three VCE engines, the non-augmented 302B provides the best range. The augmented versions showed approximately 100 n.mi. (185 km) less range. Also shown is that, for maximum range, very little augmentation is required of the augmented engines during climb.

Conclusion

Based on these results, the 302B engine has been selected for the in-depth engine-airframe integration study. NASA concurred with this selection.

ENGINE - AIRFRAME INTEGRATION

As concluded from the previous paragraph, the P&WA -302B engine is the variable cycle engine selected for in-depth airframe integration study. The data for the engine consisted of a 2.4M data package prepared by P&WA under the NASA Lewis engine study contract. DAC corrected the data to the desired conditions (2.2M and DAC installation requirements) for the integration study.

Uninstalled Performance

The uninstalled performance includes the effects of:

° U.S. 1962 model atmosphere	
° Inlet recovery	Figure 1-6
° P&WA supplied internal nozzle velocity coefficient	
° Customer compressor airbleed	1 lb/sec (.454 kg/sec)
° Customer power extraction	200 HP (149 kW)
° Jet A Fuel, Lower Heating Value	18,400 BTU/lb (4.34 x 10^7 J/kg)
° No losses for acoustical treatment	

Installed Performance Analysis

The analysis of the propulsion system performance of the -302B engine follows the same procedures used for the baseline turbojet engine (Section 1).

The inlet performance and the nacelle analysis include an evaluation of the following items:

- Inlet spillage drag
- Inlet bypass drag
- Engine and ECS cooling airflow drag
- Nacelle skin friction drag
- Nacelle afterbody drag
- Nacelle wave drag

The inlet geometry and cone schedules are the same as used for the turbojet engine. The inlet total pressure recovery variation is shown in Figure 1-6. Also shown in the figure is the variation of inlet critical mass-flow ratio. Shown in Figure 1-7 is the mass-flow ratio for the inlet boundary layer bleed airflow.

The engine airflow schedule for the -302B engine is shown in Figure 4-11. The installed inlet performance for this engine is shown in Figure 4-15. As shown by the upper graph in the figure, the inlet airflow supply provides an adequate match with the engine airflow demand. The inlet is sized at the design point of 2.2 M. The sized capture area is 29.5 sq. ft. (2.74 sq. m.). The engine and ECS cooling airflow are based on an allowance of 2.0 percent of inlet capture area airflow for the environmental control system (ECS) cooling and for engine compartment ventilation and nozzle cooling.

The nacelle drag coefficient buildup is shown in the lower graph in Figure 4-15. The inlet drag characteristics are calculated by combining the mass-flow-ratio characteristics with empirical drag coefficient correlations. For the convenience of engine sizing studies, the nacelle skin friction drag is included in the installed engine performance. The skin friction coefficients are based on fully turbulent flat plate adiabatic wall boundary layer data with transition at the leading edge and the resulting drag is shown in the figure.

The nacelle afterbody drag is dependent on the nozzle exit area and flight Mach number. The maximum nozzle area is sized at 2.2 M climb. The engine dependent boattail drag at this condition is zero. As nozzle area decreases for lower Mach numbers and reduced power settings, the boattail drag increases. The boattail drag identified with this area change is based on drag characteristics

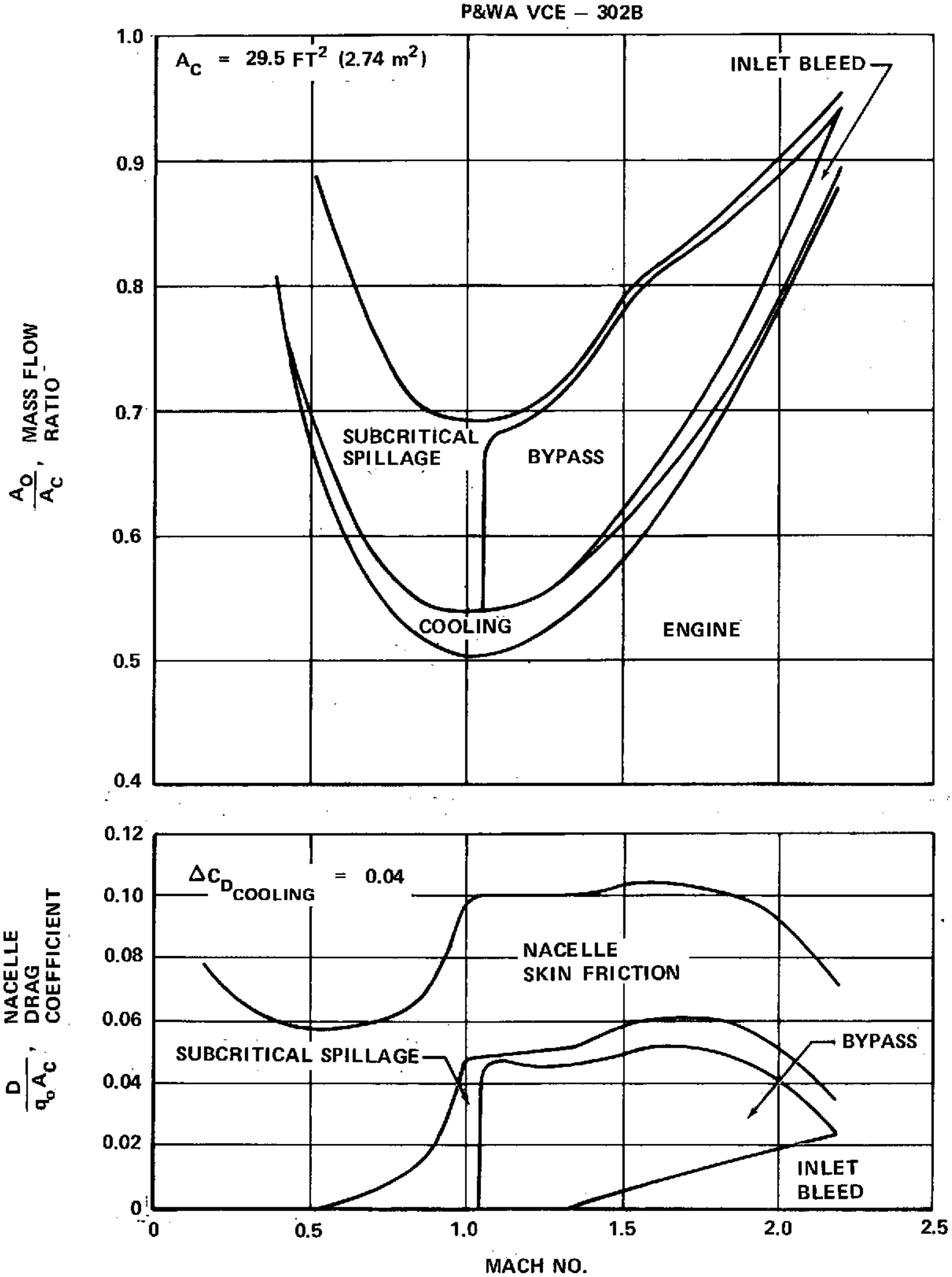


FIGURE 4-15. INSTALLED INLET PERFORMANCE

estimated for the DAC baseline configuration. The variations in drag coefficient relative to the design condition along the aircraft climb path at maximum climb thrust and for subsonic flight are shown in Figures 4-16 and 4-17.

The nacelle wave drag in the presence of the aircraft, including the supercritical spillage drag and the design afterbody drag is part of the aircraft wave drag.

Performance Results

Installed propulsion system performance is generated by correcting the uninstalled engine performance data for the installation effects described above. The climb performance characteristics are generated along the aircraft flight path shown in Figure 1-12. Uninstalled and installed thrust for the takeoff power setting are shown in Figure 4-18. Figures 4-19 and 4-20 present the uninstalled and installed thrust and SFC, respectively, for maximum climb thrust along the climb flight path. Uninstalled and installed supersonic cruise, subsonic cruise (for alternate mission), and hold performance are shown in Figures 4-21 through 4-23. Figure 4-24 presents the installed characteristics used along the descent flight path.

P&WA VCE –302B

WAT2 = 1003 LB/SEC (455 kg/SEC)
A_c = 29.5 FT^2 (2.74 m^2)
CLIMB – LOW MODE
STD DAY

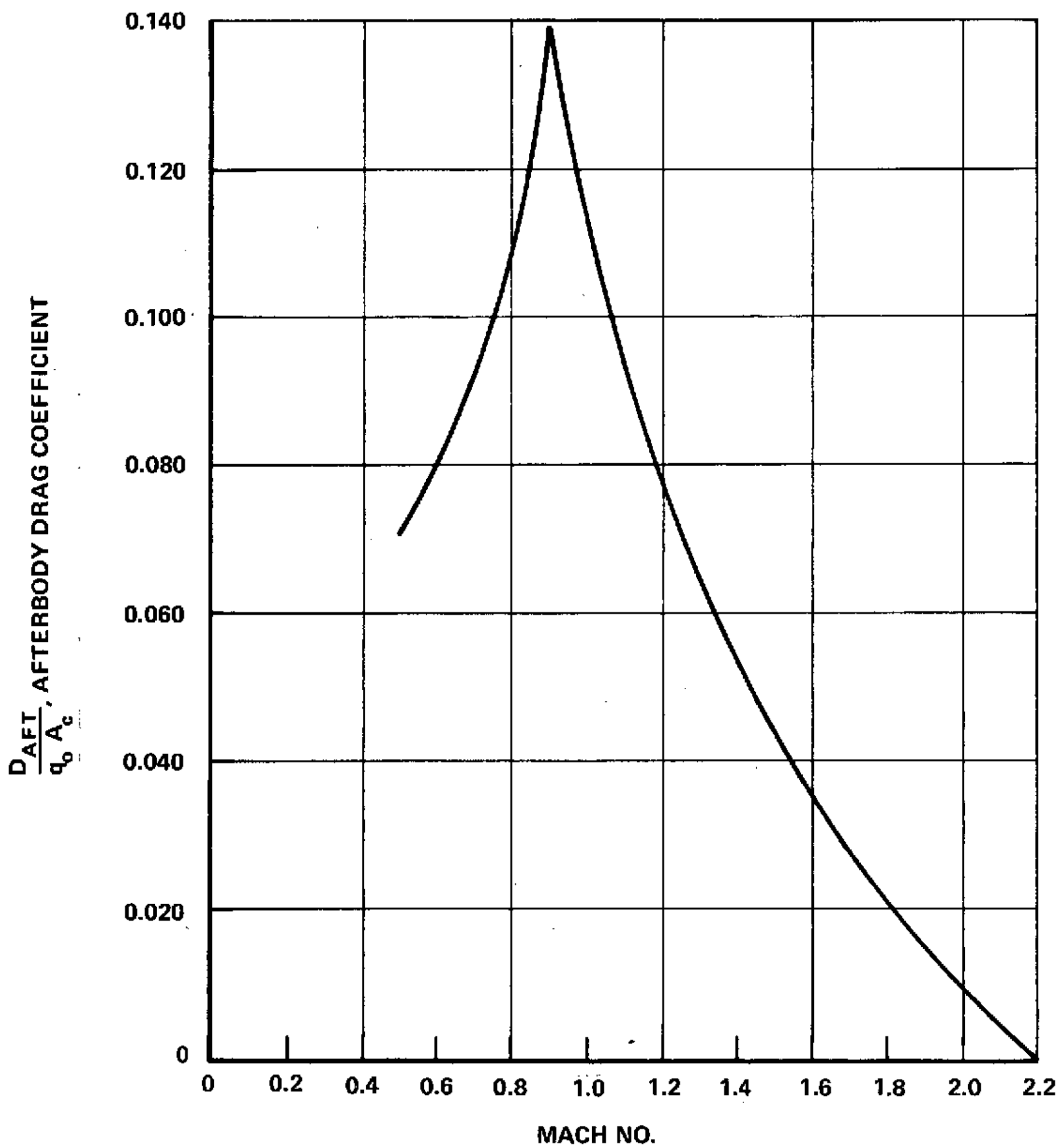


FIGURE 4-16. CLIMB AFTERBODY DRAG

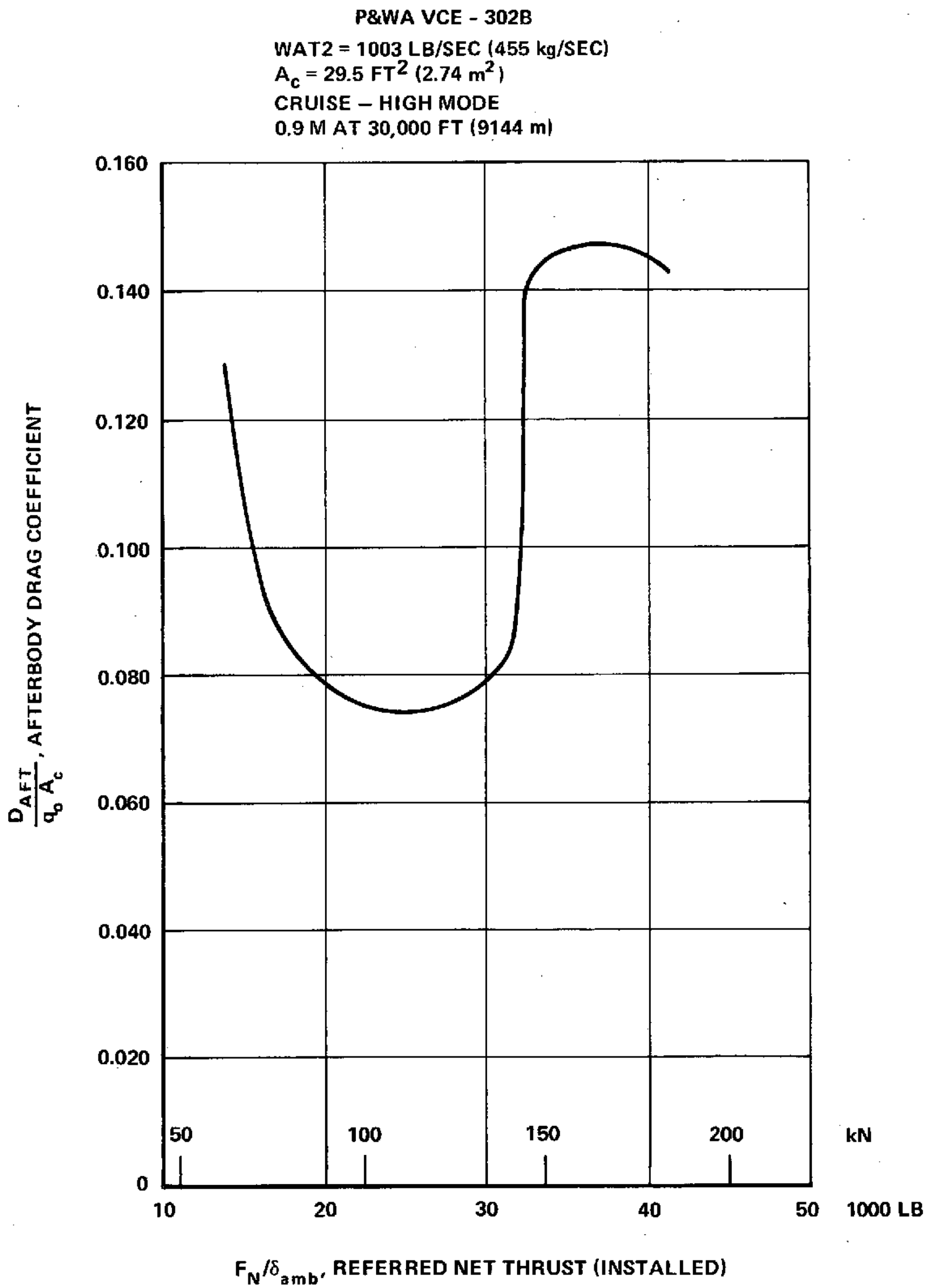


FIGURE 4-17. SUBSONIC AFTERBODY DRAG

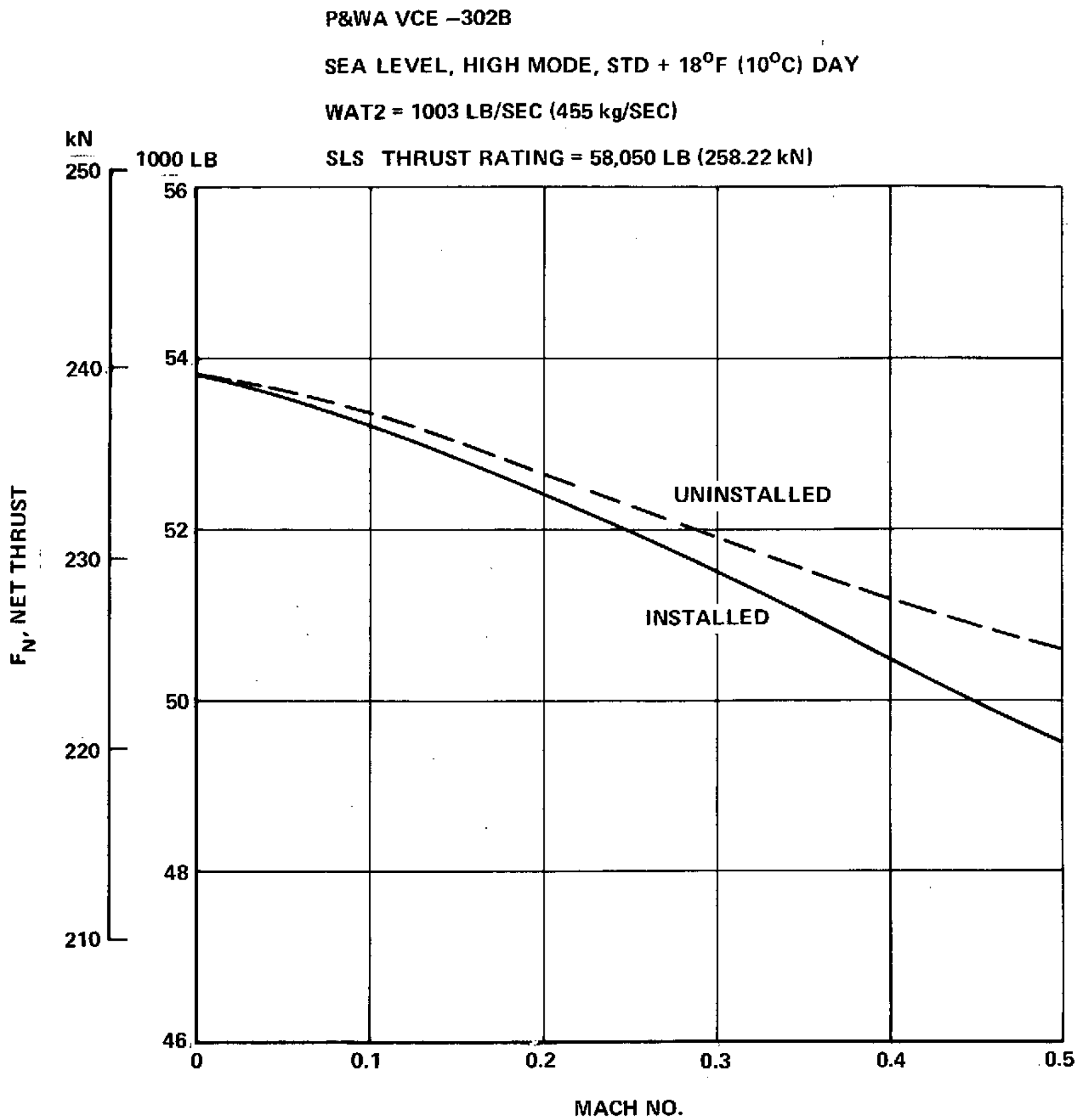


FIGURE 4-18. TAKEOFF PERFORMANCE

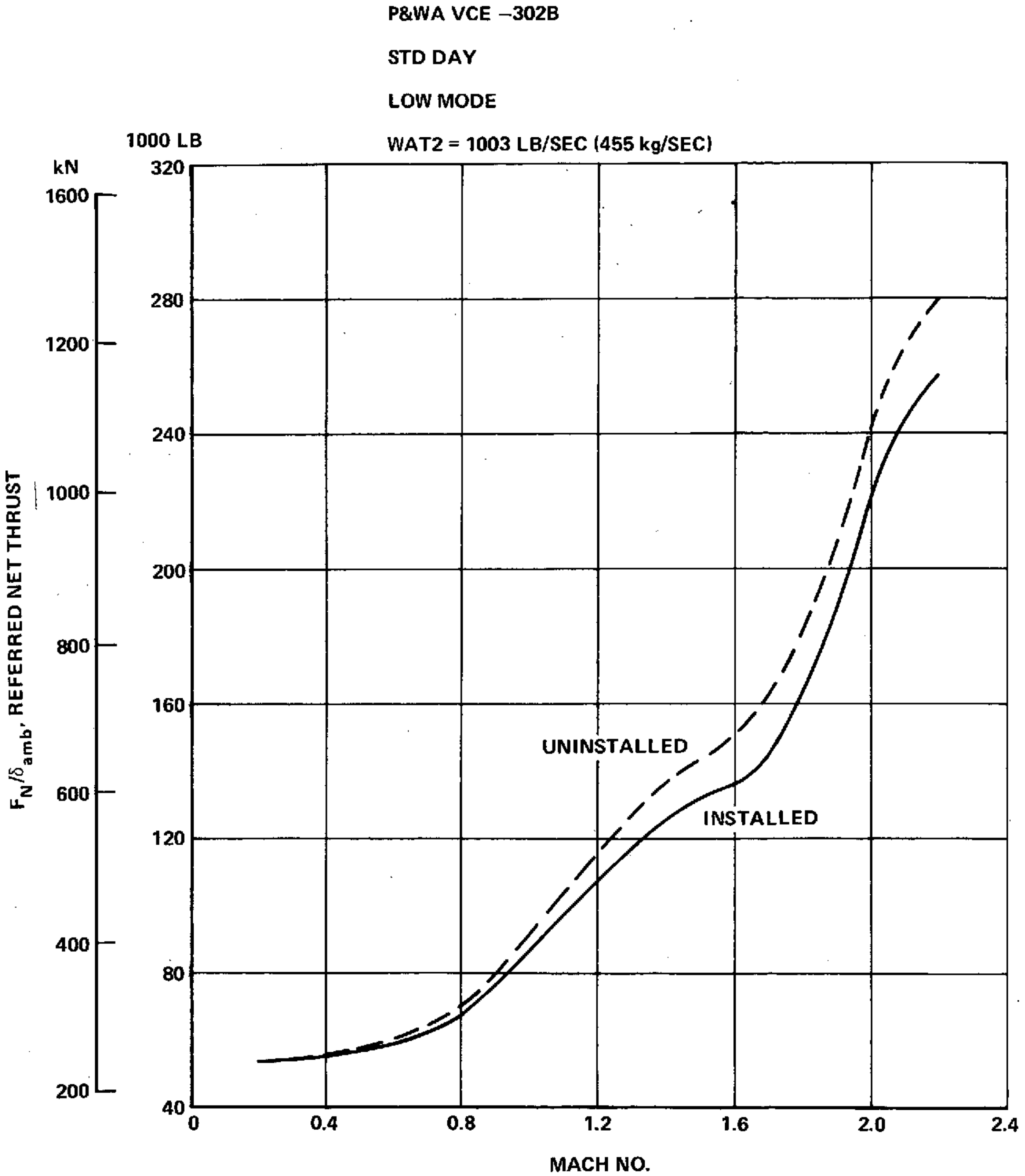


FIGURE 4-19. CLIMB THRUST

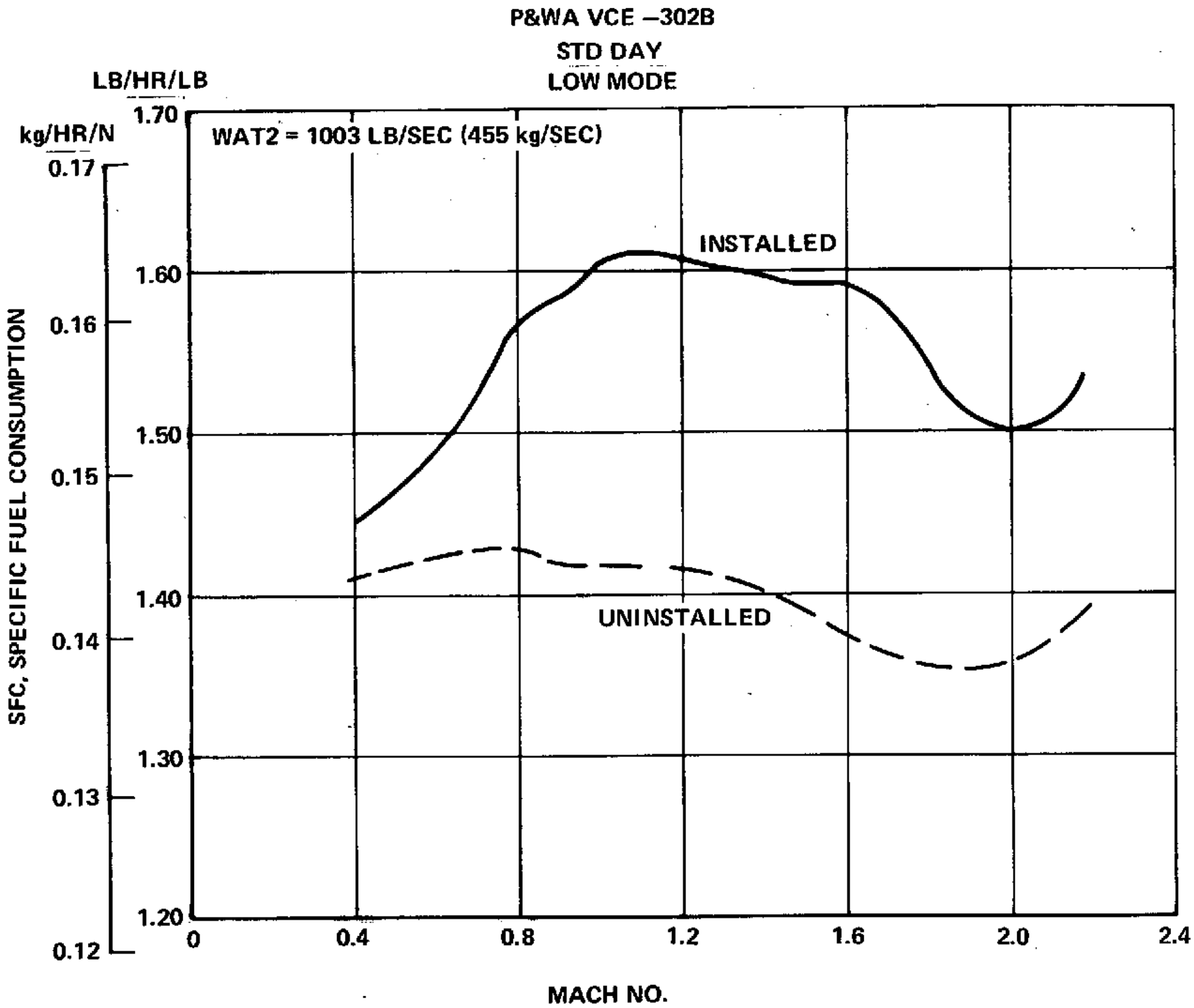


FIGURE 4-20. CLIMB SFC

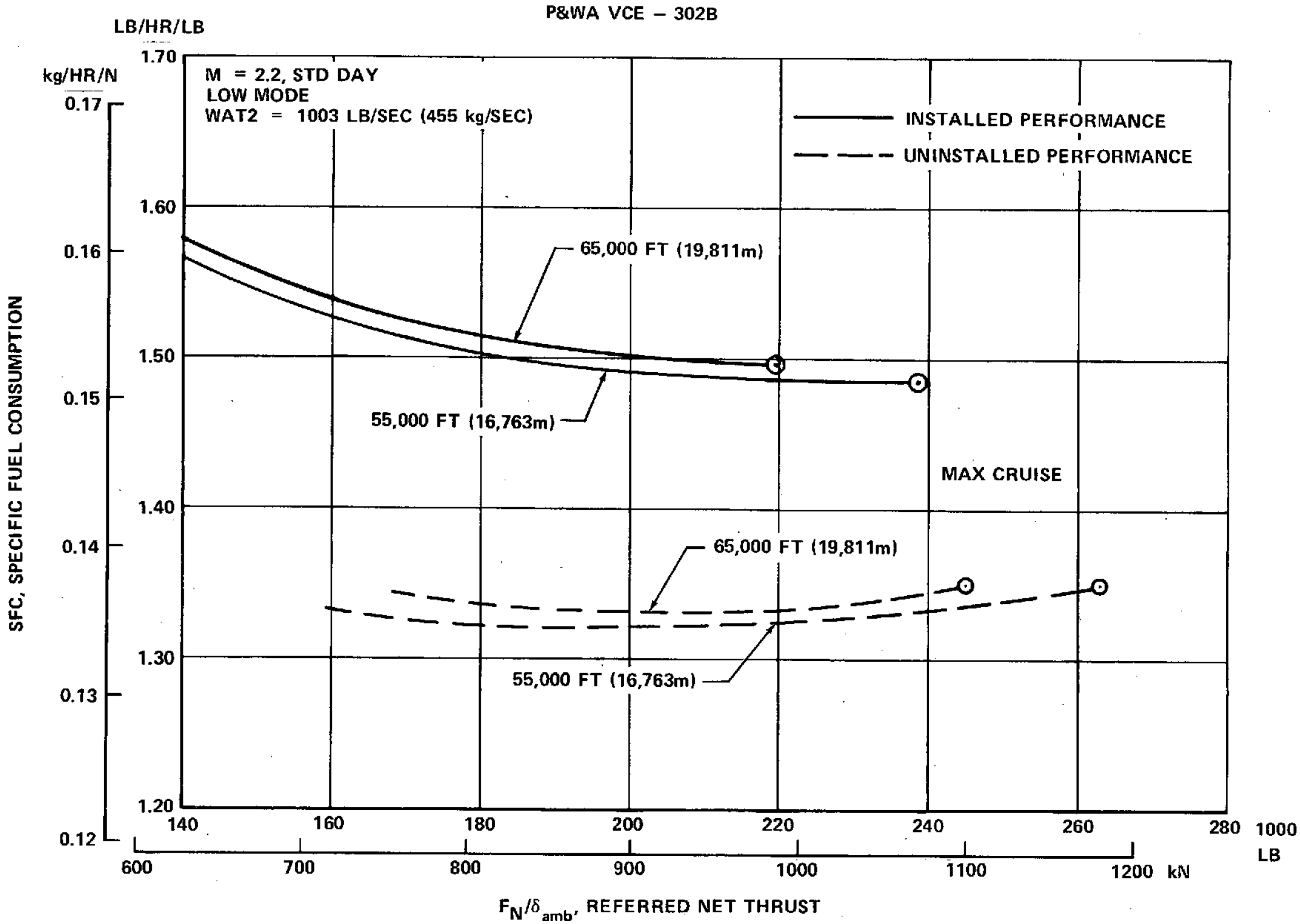


FIGURE 4-21. SUPERSONIC CRUISE PERFORMANCE

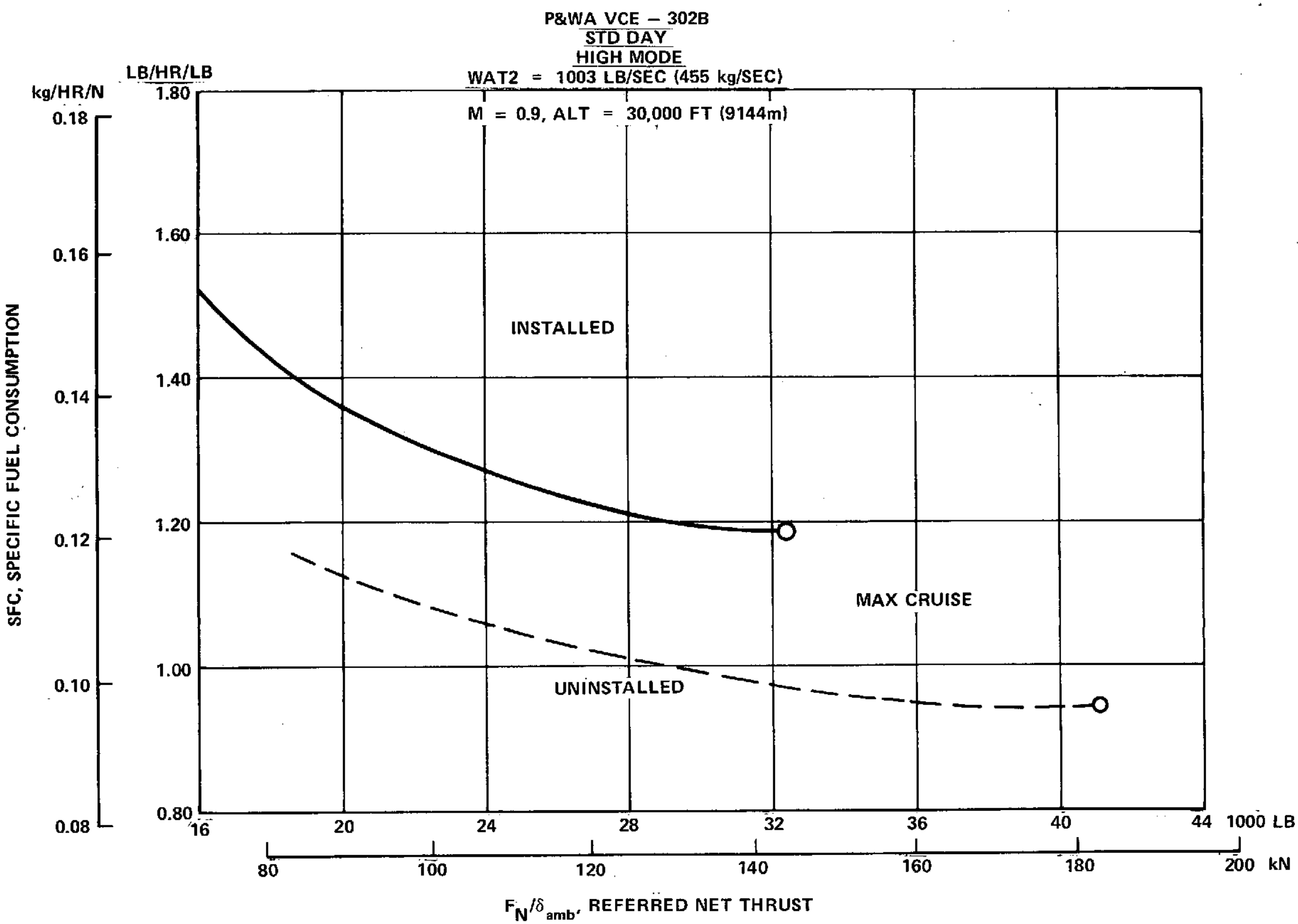


FIGURE 4-22. SUBSONIC CRUISE PERFORMANCE

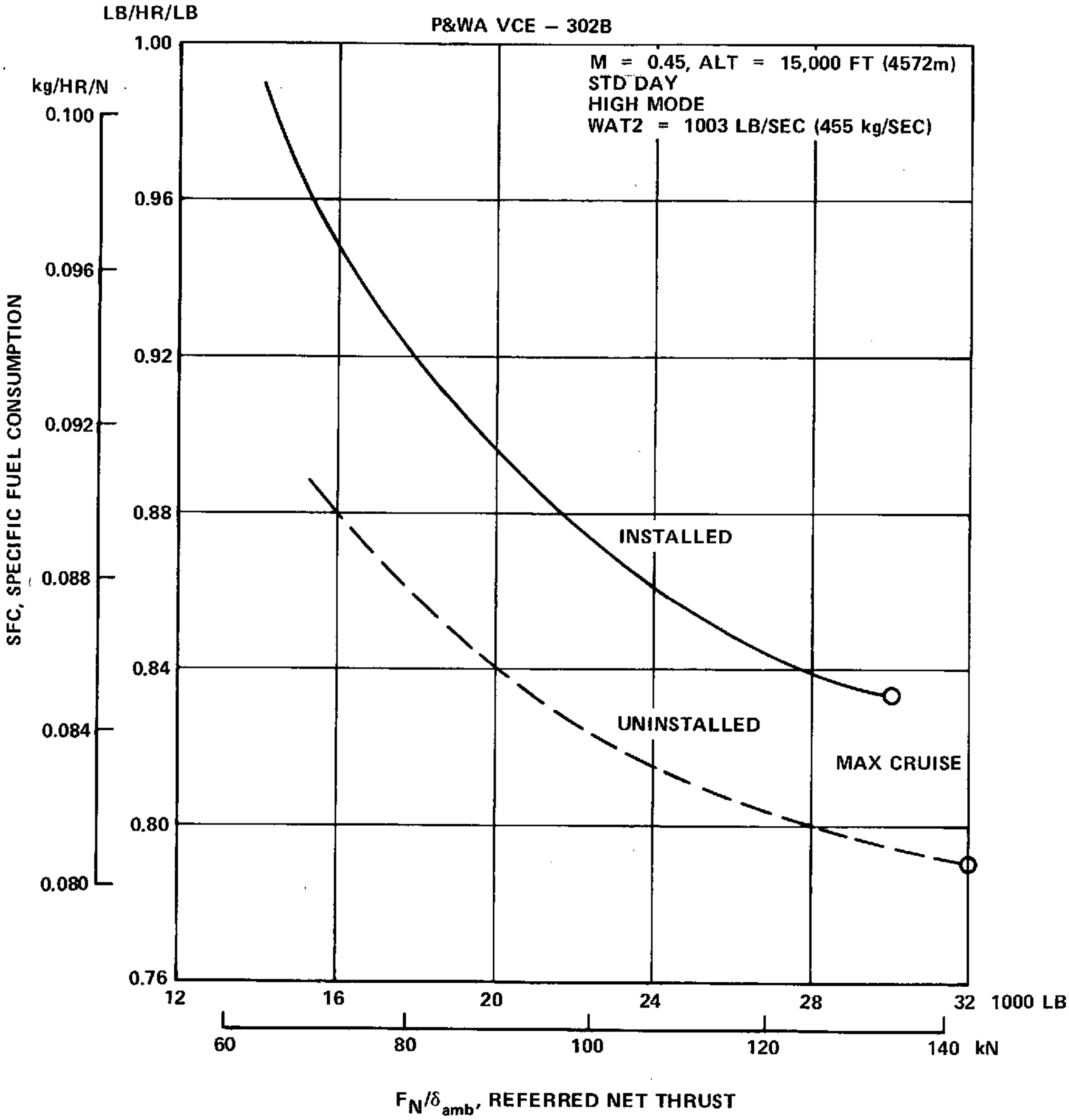


FIGURE 4-23. LOITER PERFORMANCE

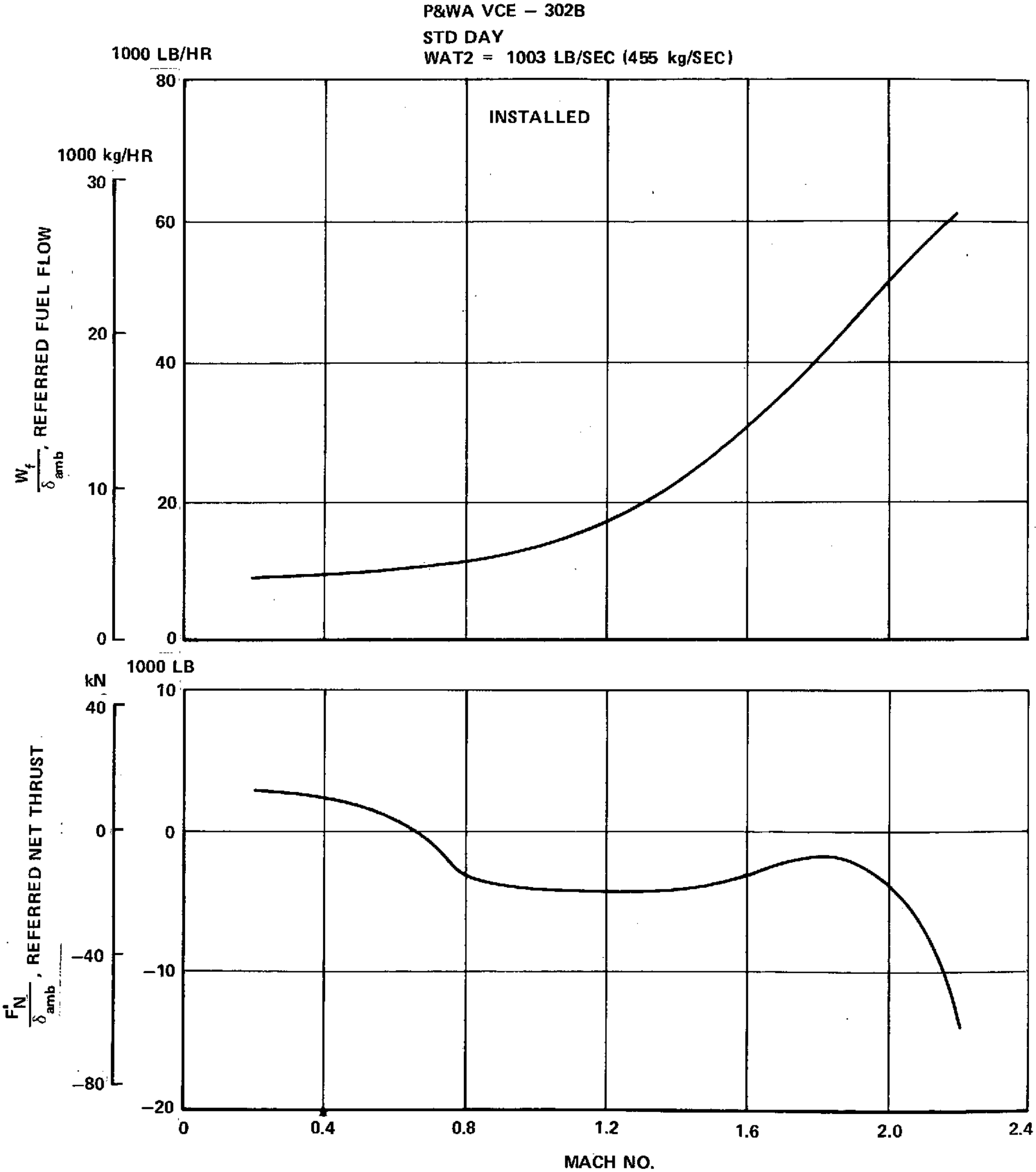


FIGURE 4-24. IDLE PERFORMANCE

Engine/Nacelle Location

Installation studies of the variable cycle dual valve engine, P&WA -302B, for the baseline airframe in four axisymmetric nacelles have been completed.

Inboard and outboard spanwise locations of engines at intake face are as for -5A, -5B and -5C aircraft.

The increase of the dimensions and weight of the -302B pod as compared to the -5A, -5B and -5C configurations make the choice of a fore and aft location of power plant to wing very critical. Aerodynamic and structural analysis of the power plant installation on the -5A, -5B and -5C configurations have established the substantial drag and/or structural penalties occurring with engine intake face and c.g. shifts both fore and aft with respect to the baseline case. This information coupled with the requirements for the mechanical attachment of the engine to the wing determined the location selected as described on the -5D configuration 3-view drawing (Figure 4-25).

The resultant position of the engine pods allows use of the total circumferential area of the nozzles for reverse thrust.

Engine Nacelle Attachment to Wing

Engine mounting to the wing is by a three-point attachment, as shown in Figure 4-26. The aft mount is on a box beam pylon cantilevered aft of the rear spar and the two forward mounts are attached to structure forward of the rear spar.

Support structure for the forward mounts is external to the wing lower surface and attached through the skin panels to the inboard and outboard pairs of slant ribs in the wing torque box. The forward right hand link from engine to wing carries thrust loads, vertical loads and side loads to the aircraft structure. The forward left hand link transmits forward and vertical loads only. The aft

CHARACTERISTICS

ITEM	WING	HORIZONTAL STABILIZER	VERTICAL STABILIZER
AREA, SQ.FT.	10000	780	700
ASPECT RATIO	1.84	2.0	0.9
TAPER RATIO	.128 (TRAP)	0.16	0.25
SWEEP OF L.E.	71° & 57°	50°	50°
DIHEDRAL	0 AT T.E.	0	~
T/C % C.	2.25-3.0	3.5	3.5
TAIL VOL. COEF.	~	.118	.044

PAYLOAD CAPACITY

MIXED CLASS

FIRST CLASS – 3 & 4 ABREAST – 38 PITCH – 40 SEATS
TOURIST CLASS – 4 5 & 6 ABREAST – 34 PITCH – 235 SEATS
TOTAL 275 SEATS

CARGO VOLUME

FORWARD	1500 CU. FT.
AFT	750 CU. FT.
TOTAL	2250 CU FT

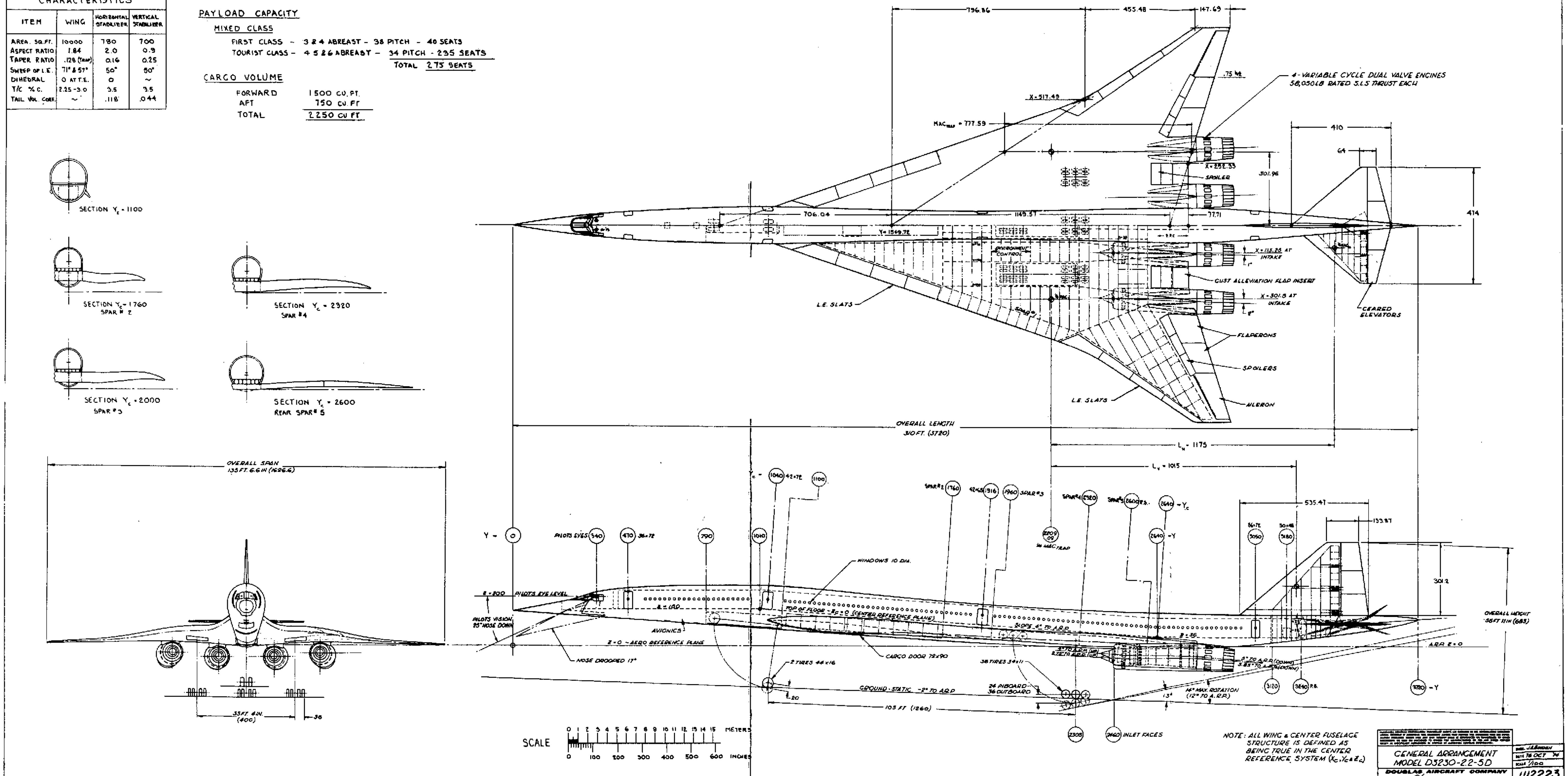


FIGURE 4-25. AST P&WA 302B VARIABLE CYCLE ENGINE CONFIGURATION

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FOLDOUT FRAME 2

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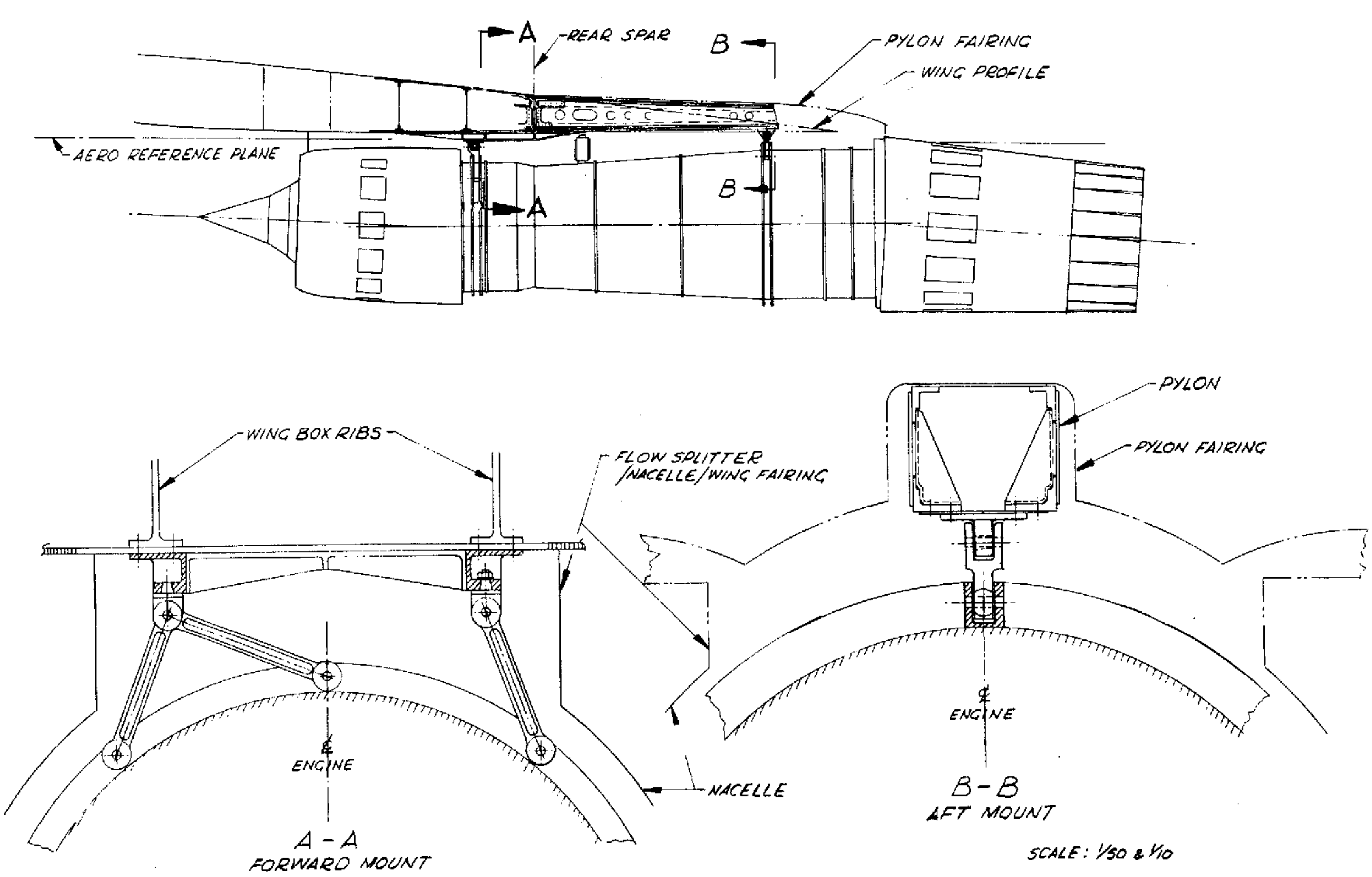


FIGURE 4-26. P&WA VCE 302B INSTALLATION SCHEMATIC

mounting link carries vertical loads and translates for engine growth under operating temperatures (Figure 4-26). Due to the size and weight of the -302B engine, the support pylon protrudes above the upper surface of the wing aft of the rear spar. These protruberances are shown on the three view drawing -5D configuration, fairing from the rear spar aft to the engine nozzle forward edge on the upper ℄ of engine.

The axisymmetric intakes are attached to the engine casing and divorced from the wing structure. This prevents transmission of wing deflection loads and associated distortion of intake geometry or loading of engine casing.

The boundary layer diverter is integrated into the engine nacelle/wing fairing.

<u>Other Configuration Changes</u>

The location and size of the VCE engine pods dictate an increase of 26 inches (66 cm) in the length of the landing gear struts to maintain ground clearance to engine nozzles on maximum rotation. To accommodate this increased length of gear in the wheel well, spar number 3 was relocated at Sta. Y_c = 1960 (was Y_c = 2000). The forward nose gear bulkhead is relocated to Sta. Y = 790 (was Y = 810). Main and nose landing gear doors are lengthened to suit.

Provision for the environment control bay is relocated between Sta. Y_c = 1960 and Y_c = 2120. As the engine nozzles become the critical point for aircraft ground clearance on rotation, the tail bumper and ventral fairing are deleted. Clearance of rear fuselage to ground on 14° maximum rotation is 55 inches (139.7 cm).

The relocation of spar number 3 necessitates an adjustment of the fuel tank arrangement.

Tanks No. 1 and No. 4 main, No. 2 and No. 3 main, and No. 2 and No. 3 alternate are resized which produces an overall reduction in fuel capacity of 6,700 lb. (304 kg) per aircraft.

ACOUSTIC ANALYSES

The acoustic analysis conducted for the variable cycle engine powered aircraft configuration consists of the calculation of jet noise estimates in support of engine sizing studies. Engine data are employed to estimate the jet noise at aircraft Mach numbers and altitudes representative of the FAR Part 36 takeoff and sideline measuring conditions. The standard climb profile features a thrust cutback over the takeoff measuring station.

The engine size is selected at an airflow rate of 1003 lbs./sec (455 kg/sec) with no jet noise suppression required. A description of the engine sizing analysis is given in the Engine Sizing section.

The predicted unsuppressed jet noise levels for the 302B engine in the baseline airplane based on specific engine conditions along the calculated takeoff trajectory are as follows:

FAR PART 36 MEASURING STATION	DISTANCE, FT.	UNSUPPRESSED TOTAL NOISE, EPNL, EPNdB
Sideline	2270 (747 m)	106.5*
Takeoff/with Cutback	1225 (374 m)	106.3

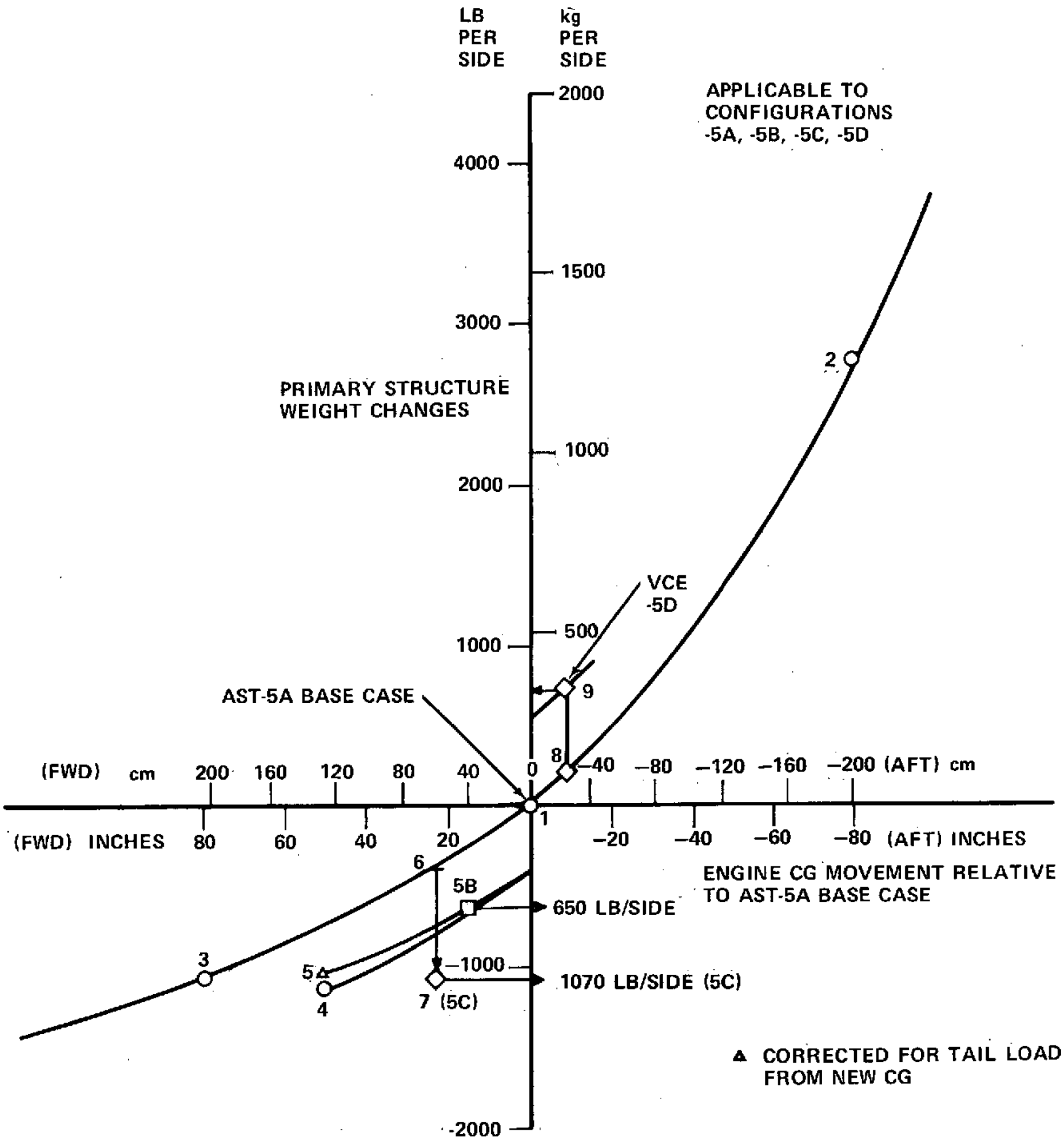
*Includes allowance of 3.0 EPNdB for extra ground attenuation.

Strength Analysis

The engine propulsion plus nacelle weight from Table 4-5 is 99,417 pounds (45,095 kg) with its c.g. located nine inches (23 cm) aft of the -5A baseline configuration. The nine inch (23 cm) aft movement would increase the structural optimization weight by 200 pounds (91 kg) per side (see point 8 on Figure 4-27). The increase in propulsion system plus inlet weight of 7249 pounds (3288 kg) per side (Table 4-5) over the -5A baseline results in a structural optimization weight increase of 537 pounds (244 kg) per side (see Figure 4-27, point 9 and point 8). The sum for the aft c.g. location and increased weight is 737 pounds (334 kg) per side or 1474 pounds (669 kg) per airplane.

Flutter Analysis

The baseline case described in Section 2 has been analyzed further by strength optimizing the structure, increasing the stiffness for roll and control effectiveness, and including fuel in the outer wing tank. Using this base, the -5C (duct heating turbofan), and -5D (VCE) have been analyzed for incremental flutter effects. The -5D zero speed bending frequency is lower than the torsion frequency and rises more slowly with airspeed because of its heavier mass, thus causing coalescence with the flat rising torsion mode at a slightly higher speed. The opposite occurs with the lighter weight, -5C. Flutter occurs at a slightly lower speed than the baseline case, which is 450 KEAS (.232 m/sec EAS) at 100 percent fuel. The lower flutter speed for -5C, 425 KEAS (.218 m/sec EAS) is approximately the same as an earlier non-optimized base case, -5A-1. Adding 860 lbs. (390 kg) for flutter optimization for case -5A-1 was sufficient to increase this speed to the required flutter speed of 480 KEAS (.247 m/sec EAS). It is anticipated that an 860 lb. (390 kg) weight addition for flutter optimization (see Table 4-5) with the option for fuel programming, that is, including fuel in the outer tank, will provide the



PTS 1-5 = STRUCTURAL ANALYSIS POINTS

Δ= TAIL LOAD CORRECTION

6 = WGT SAVING FOR A 23.5-INCH FORWARD CG FOR DUCT HEATER (5C)

7 (5C) = LOCATION PLUS PROPULSION SYSTEM WEIGHT REDUCTION FOR DUCT HEATER (5C)

FIGURE 4-27. STRUCTURAL WEIGHT CHANGE FOR ENGINE LOCATION AND SIZE required flutter speed of 480 KEAS (.247 m/sec EAS) for all configurations analyzed in this report.

Table 4-5 compares the weight of the AST, with dual cycle P&WA -302B engines (-5D), to the turbojet baseline (-5A). The engines are scaled to a corrected air flow of 1,003 lbs/sec (455 kg/sec). Each engine installation weighs 19,575 lbs. (8879 kg), including 3,712 lbs. (1684 kg) for nozzle and reverser. Comparable weights for the -5A baseline are 16,982 lbs. (7703 kg) and 4,040 lbs. (1833 kg) respectively. Total propulsion system weight is 80,400 lbs. (36,469 kg), 10,210 lbs. (4631 kg) greater than the -5A baseline turbojet, 20,469 lbs. (9285 kg) greater than the mini-bypass -5B and 29,343 lbs. (13,310 kg) greater than the duct heating turbofan, -5C.

The nacelle/inlet is 4,287 lbs. (1945 kg) heavier than the baseline, -5A. About half of this, 2,295 lbs. (1041 kg), reflects an increase in the weight of the engine cowling. This is due to a larger engine envelope. The remainder, 1,992 lbs. (904 kg), reflects a heavier inlet installation, the result of an increase in both length and capture area.

The 1,474 lbs. (669 kg) estimated for Structural Weight Increment accounts for differences in pylon and engine support weight, along with differences in wing and fuselage weight due to changes in load. The weight estimating approach is discussed in Section 1. Analysis and derivation of the weight penalty for aeroelasticity and flutter is discussed in the previous paragraph of this section.

Minimum ground to exhaust nozzle clearance, at maximum rotation, establishes the length of the main gear strut. The geometry and location of the -302B engine necessitates a 26 inch (66 cm) increase in the length of the main gear strut. An equivalent increase is required in the length of the nose gear strut, to retain the same fuselage attitude during ground operation. This results in a 1,220 lb. (553 kg) increase in gear weight, plus an additional

TABLE 4-5.

WEIGHT COMPARISON – CONFIGURATION 5D (VARIABLE CYCLE ENGINE) WITH 5A BASELINE (TURBOJET) ENGLISH UNITS

CONFIGURATION	WEIGHT – POUNDS		
	5A TURBOJET	5D VCE	DIFF.
ITEM			
WING	75,347	75,537*	+190
H-TAIL	3,960	3,960*	0
V-TAIL	3,807	3,807*	0
FUSELAGE	47,713	47,762*	+49
LANDING GEAR	36,792	38,012	+1,220
FLIGHT CONTROLS	9,115	9,115	0
NACELLE/INLET	14,730	19,017	+4,287
PROPULSION (LESS FUEL SYSTEM)	70,190	80,400	+10,210
FUEL SYSTEM	3,820	3,820	0
EMERGENCY POWER UNIT	950	950	0
INSTRUMENTS	1,227	1,227	0
HYDRAULICS	5,684	5,684	0
PNEUMATICS	1,332	1,332	0
ELECTRICAL	4,850	4,850	0
NAVIGATION AND COMMUNICATIONS SYSTEM	2,756	2,756	0
FURNISHINGS	24,478	24,478	0
AIR CONDITIONING	4,854	4,854	0
ICE PROTECTION	489	489	0
HANDLING PROVISIONS	90	90	0
PENALTY – FLUTTER AND AEROELASTICITY	2,860**	2,860**	0
STRUCTURAL WEIGHT INCREMENT		1,474*	+1,474
MANUFACTURER'S EMPTY WEIGHT	315,044	332,474	+17,430
OPERATIONAL ITEMS	8,096	8,096	0
OPERATIONAL EMPTY WEIGHT	323,140	340,570	+17,430

*THE WEIGHT INCREMENT FOR STRENGTH, ETC., FOR THESE ITEMS IS INCLUDED UNDER THE ITEM STRUCTURAL WEIGHT INCREMENT AND LISTED SEPARATELY.

**2000 LB FOR ROLL AND CONTROL EFFECTIVENESS
860 LB FOR FLUTTER OPTIMIZATION

TABLE 4-5.

WEIGHT COMPARISON – CONFIGURATION 5D
(VARIABLE CYCLE ENGINE) WITH 5A BASELINE (TURBOJET)
METRIC UNITS

CONFIGURATION	WEIGHT – KILOGRAMS		
	5A TURBOJET	5D VCE	DIFF.
ITEM			
WING	34,177	34,263*	+86
H-TAIL	1,796	1,796*	0
V-TAIL	1,727	1,727*	0
FUSELAGE	21,642	21,664*	+22
LANDING GEAR	16,689	17,242	+553
FLIGHT CONTROLS	4,134	4,134	0
NACELLE/INLET	6,681	8,626	+1,945
PROPULSION (LESS FUEL SYSTEM)	31,838	36,469	+4,631
FUEL SYSTEM	1,733	1,733	0
EMERGENCY POWER UNIT	431	431	0
INSTRUMENTS	557	557	0
HYDRAULICS	2,578	2,578	0
PNEUMATICS	604	604	0
ELECTRICAL	2,200	2,200	0
NAVIGATION AND COMMUNICATIONS SYSTEM	1,250	1,250	0
FURNISHINGS	11,103	11,103	0
AIR CONDITIONING	2,202	2,202	0
ICE PROTECTION	222	222	0
HANDLING PROVISIONS	41	41	0
PENALTY – FLUTTER AND AEROELASTICITY	1,297	1,297	0
STRUCTURAL WEIGHT INCREMENT	—	669*	+669
MANUFACTURER'S EMPTY WEIGHT	142,902	150,808	+7,906
OPERATIONAL ITEMS	3,672	3,672	0
OPERATIONAL EMPTY WEIGHT	146,574	154,480	+7,906

*THE WEIGHT INCREMENT FOR STRENGTH, ETC., FOR THESE ITEMS IS INCLUDED UNDER THE ITEM STRUCTURAL WEIGHT INCREMENT AND LISTED SEPARATELY.

239 lb. (108 kg) increase in wing and fuselage structure, to accommodate the longer struts.

The location of the mean line of the inlets of the variable cycle engines is at Sta. 2460. This is 40 inches (101 cm) forward as compared to the turbojet baseline -5A. The c.g. of the engine installation, however, is at station 2694, 9 inches (23 cm) aft of the baseline, -5A. The combined effect of the shift in engine c.g., increased engine installation weight and heavier gear and structure, moves the operational empty weight c.g. 21.5 inches (54.5 cm) aft of the baseline configuration. The total increase in operational empty weight is 17,430 lbs. (7906 kg). Total OEW including this increase is 340,570 lbs. (154,480 kg).

Aerodynamics Analysis

The trimmed lift and drag characteristics for the -302B powered aircraft are obtained by adjusting the wave drag of the baseline turbojet powered aircraft for the difference due to the larger nacelles. The difference in nacelle skin friction drag is accounted for in the installed propulsion system performance. The wave drag program predicts a reduction in supersonic wave drag of 1.59 counts (ΔC_D = .000159) due to the differences in nacelle shape and location. The characteristics used to determine the mission performance for the -302B powered aircraft are obtained by subtracting this increment from the wave drag of the baseline turbojet powered aircraft, -5A.

Performance Results

Estimated performance characteristics for the -302B powered aircraft are presented in Figures 4-28 through 4-30 as a function of engine size. The mission profile and reserve ground rules are the same as used for the baseline turbojet aircraft (Figure 1-20). The takeoff gross weight is held constant at 750,000 lb. (340,194 kg) and the payload is fixed at 55,965 lb. (25,385 kg).

Figure 4-28 presents the takeoff characteristics and the height above the runway at 3.5 n.mi. (6.5 km) from the start of takeoff, with the throttle cut back to meet the 4 percent all engine climb gradient requirement of FAR Part 36. The characteristics of the aircraft with the engine size selected as described in the engine sizing section are indicated on the figure. The performance of the baseline turbojet aircraft, -5A, is shown for reference.

Figure 4-29 presents the variation of operator's weight empty with engine size used for the mission performance calculations, the altitude for maximum range factor at the start of the 2.2 M cruise, and the mission range.

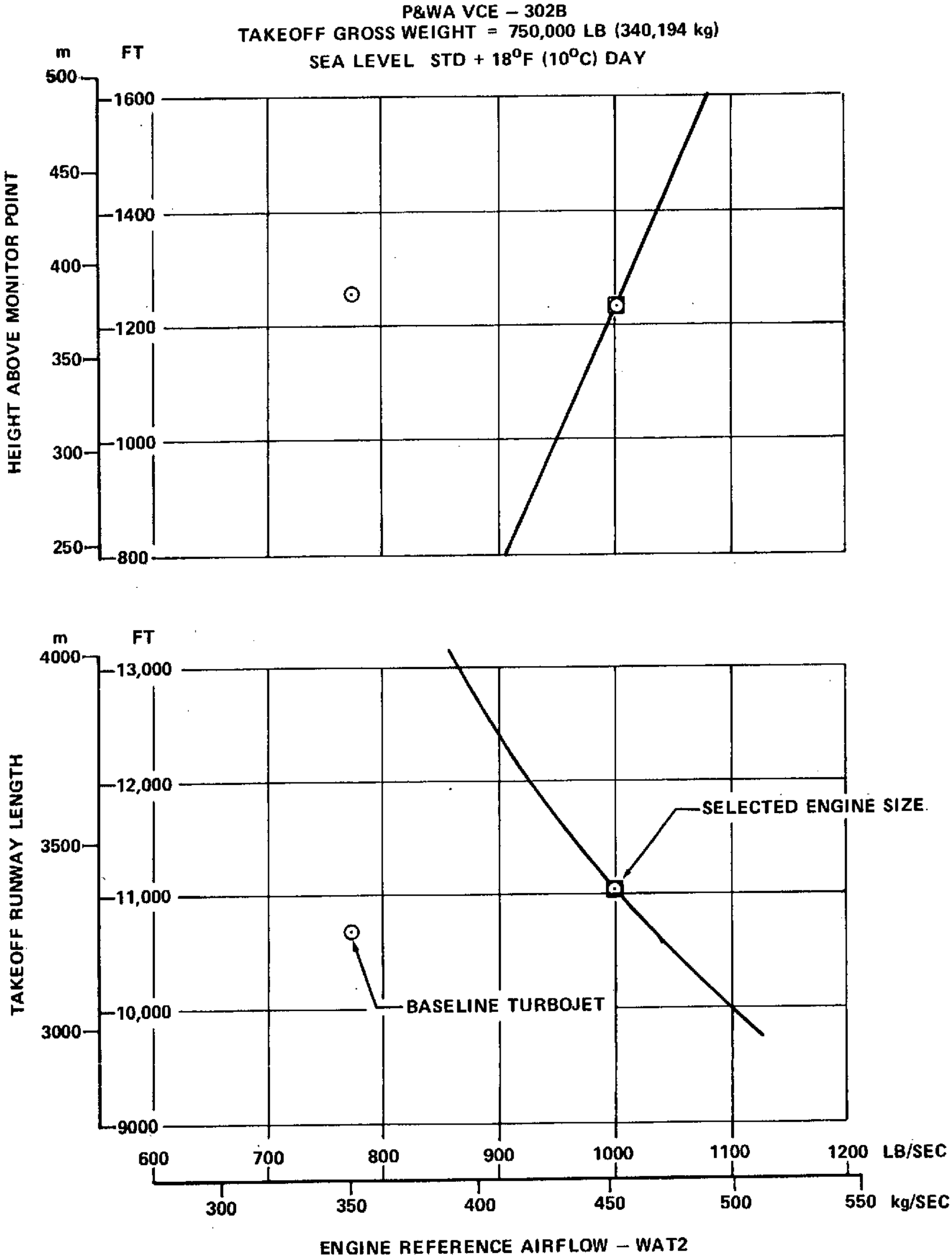


FIGURE 4-28. EFFECT OF ENGINE SIZE ON TAKEOFF PERFORMANCE

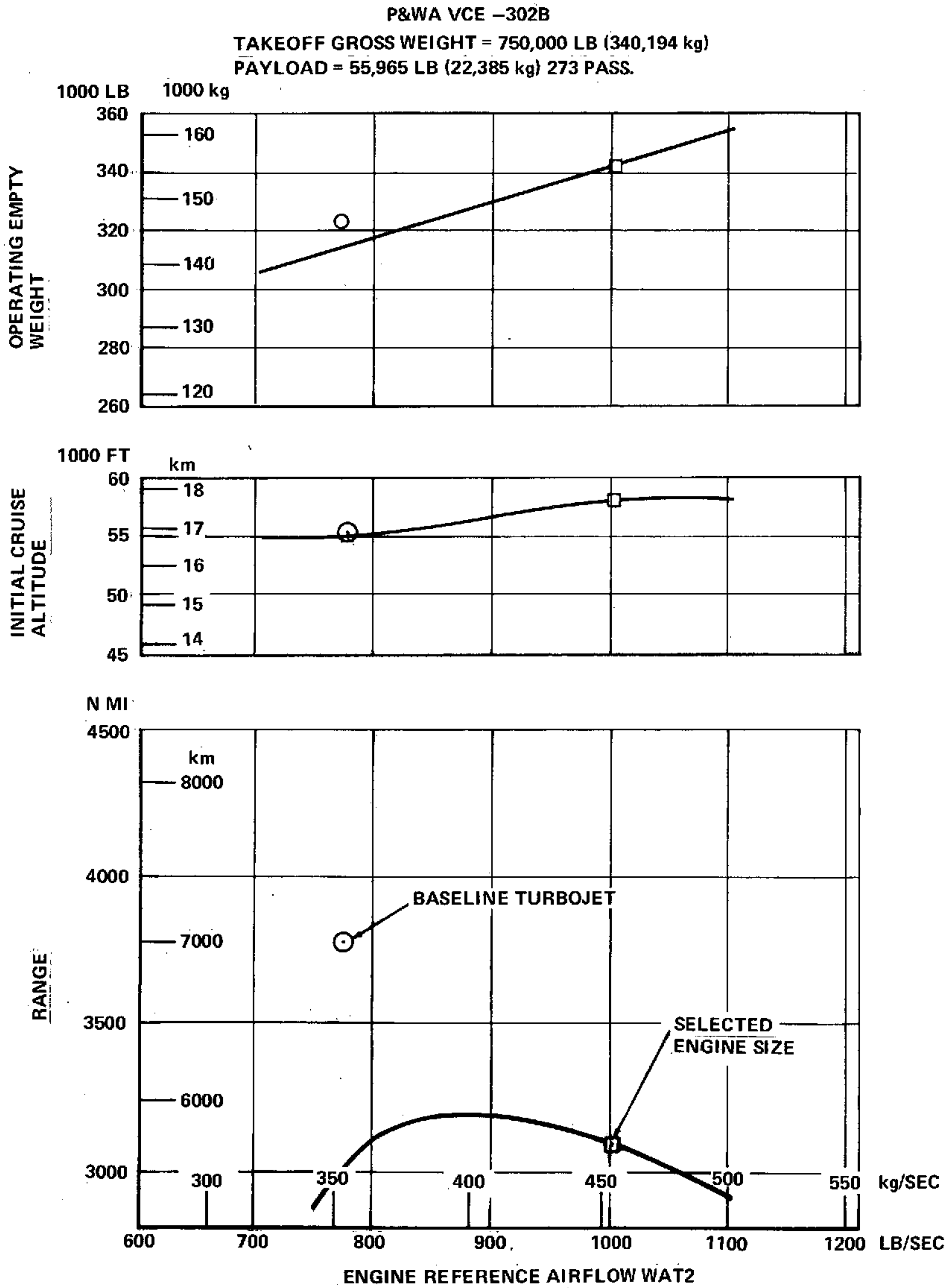


FIGURE 4-29. EFFECT OF ENGINE SIZE ON MISSION PERFORMANCE

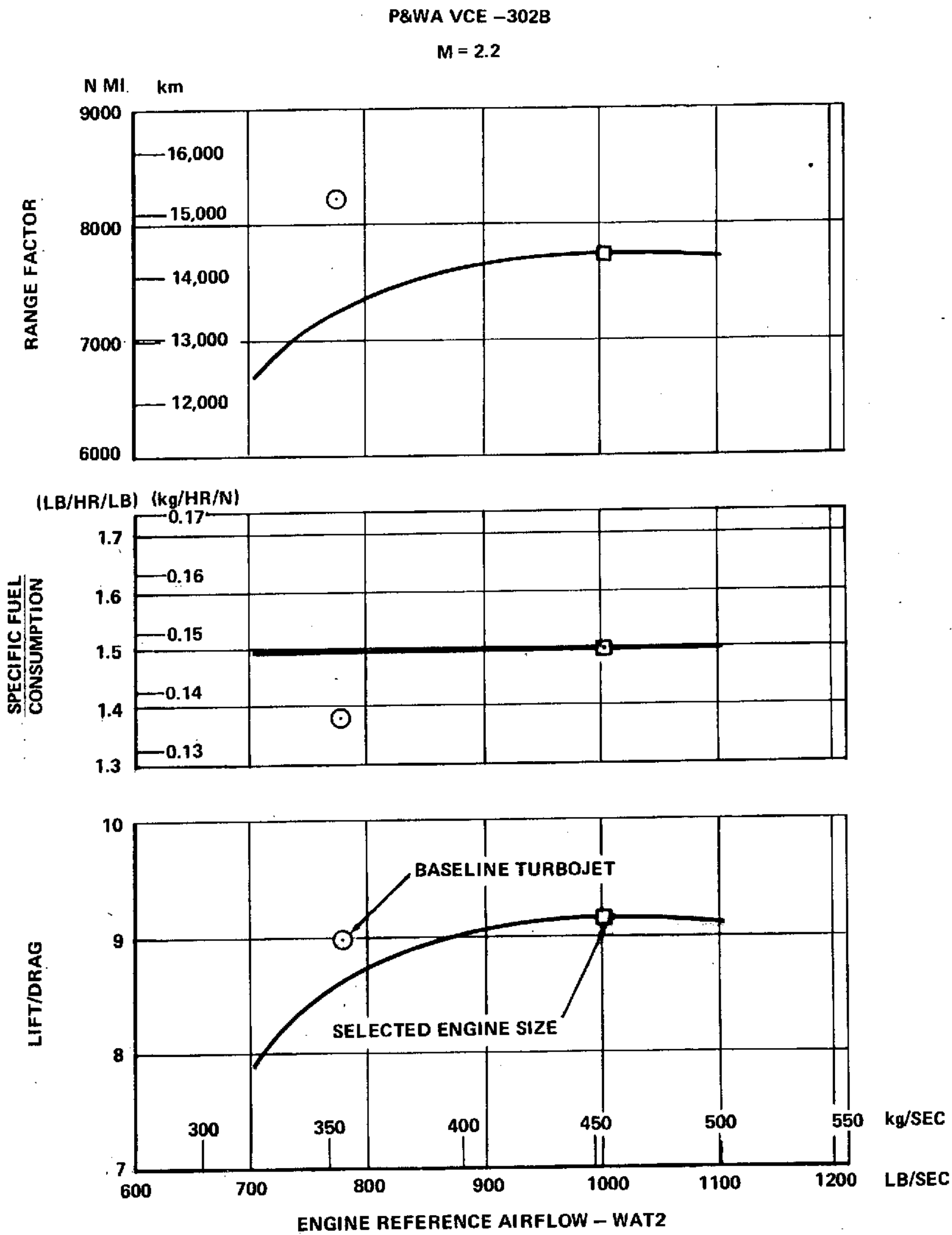


FIGURE 4-30. EFFECT OF ENGINE SIZE ON CRUISE PARAMETERS

The selected engine size as indicated in the figure is identical to that specified in the engine sizing paragraph. A smaller size engine indicates better range, however, the -302B is sized at its maximum takeoff thrust, unthrottled, with no suppressor and cannot be sized smaller without significant penalty in takeoff performance. Figure 4-30 presents some of the details of the effect of engine size on the optimum cruise L/D, cruise installed SFC, and the 2.2 M cruise range factor.

For changes with engine sizing, the data presented in Figures 4-29 and 4-30 account for the engine and nacelle weight, and inlet and nacelle drag, but neglects the changes in aircraft wave drag. For a ten percent change in engine size, this wave drag effect on performance is quite small, but can be significant for the larger engine sizes.

The performance for the -302B powered aircraft with the 1003 lb/sec (455 kg/sec) engine is summarized below:

Takeoff Gross Weight	750,000 lb. (340, 194 kg)
Payload	55,965 lb. (25,385 kg)
Takeoff Field Length	11,000 ft. (3350 m)
Height at 3.5 n.mi. (6.5 km) Takeoff Point	1,225 ft. (373 m)
Range	3,090 n.mi. (5722 km)
Initial Cruise Altitude	58,060 ft. (17.7 km)
Direct Operating Cost (1973 $)	2.21 cents/seat n.mi.

The variation in range vs. initial subsonic leg length is shown in Figure 4-31. For a 600 n.mi. initial subsonic leg, the range penalty is 3 percent.

P&WA VCE −302B

TAKEOFF GROSS WEIGHT = 750,000 LB (340,194 kg)
PAYLOAD = 55,965 LB (22,385 kg) 273 PASS.

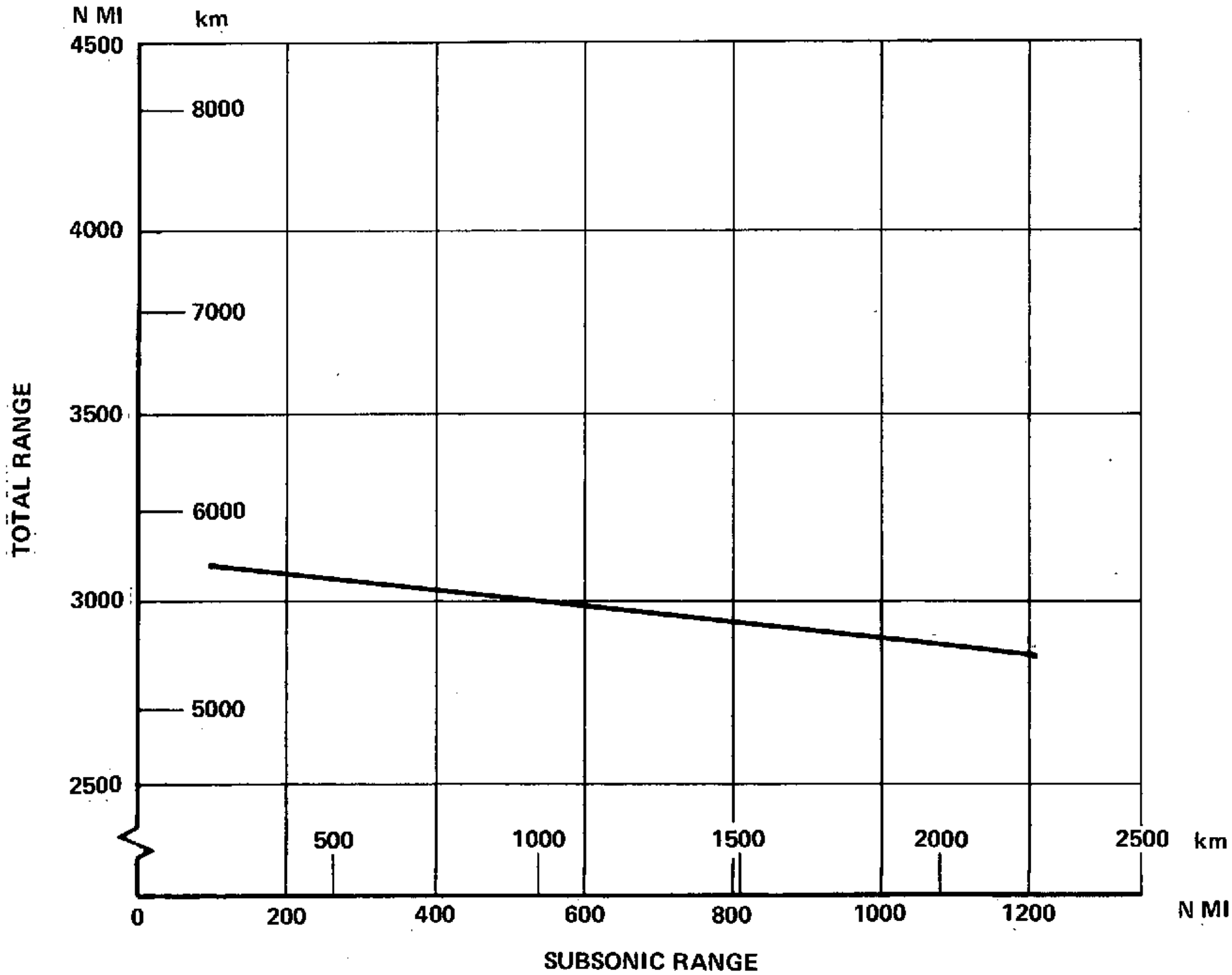


FIGURE 4-31. EFFECT OF INITIAL SUBSONIC LEG ON RANGE